UNIVERSITAT POLITÈCNICA DE CATALUNYA
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# TREBALL DE FI DE GRAU

**TFG TITLE: Satellite Mission Analysis Simulator for Earth Observation**

**DEGREE: Bachelor's Degree in Aerospace Systems Engineering**

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**DATE: June 17, 2019**

**Títol:** Simulador d'anàlisi de missió en satèl·lit per l'observació de la Terra

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## Resum


El principal objectiu d'aquest treball de fi de grau es estudiar i millorar un software de missió espacial amb satèl·lit desenvolupat amb Matlab per a l'observació de la Terra

Primer, las caracteristicas i l'estructra del software existent es analitzat de manera que tot es correctament entès, per tal de definir les millores apropiades.
Aquestes millores estan aplicades al propagador orbital, als diferents subsistemes del satèl·lit com per exemple el de potència, el tèrmic o el de dates, als modes i a altres conceptes importants sobre una missió amb satèl·lit, de manera que el simulador és el més aproximat possible a un escenari real de missió.

Un cop s'han implementat i validat les millores de software, un exemple d'anàlisi de missió real amb satèl·lit es elaborat. Aquesta missió consisteix a observar una zona geogràfica de la Terra amb uns objectius i restriccions...

Finalment, el resultat del projecte és una aplicació GUIDE a partir del software de Matlab per tal que es puguin programar i estudiar diferents missions. Analitzant els objectius d'una determinada missió juntament amb les restriccions, l'usuari pot provar diferents combinacions entre el disseny del satèl·lit, l'òrbita, modes, estacions de terra... i realitzar una comparació entre totes aquestes combinacions per obtenir el millor resultat possible per aconseguir l'objectiu.

La metodologia utilitzada per desenvolupar el projecte, primerament es desenvolupar un estudi dels paràmetres típics d'una missió espacial i dels conceptes relacionats amb satèl·lits a partir de llibres i articles de recerca, per tal d'aconseguir una bona base teòrica. Amb aquests nous coneixements adquirits, el software existent pot ser entès i millorat mitjançant la implementació d'aquests nous conceptes, de la millor manera possible.

La validació del projecte es realitza comparant els resultats de cada nova implementació, tant individualment com en conjunt, amb la primera versió del simulador i la base teòrica estudiada per comprovar la fiabilitat del software.

**Title :** Small Satellite Mission Analysis Simulator for Earth Observation

**Author:** Álvaro Crespo Serrano

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Jordi Castellví

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## Overview

The aim of this final degree project is to study and improve an existing satellite mission simulator software developed using Matlab for Earth observation purposes.

Firstly, the characteristics and structure of the existing software is analyzed so that everything is properly understood in order to define appropriate improvements.
These improvements are applied to the orbital propagator, the different subsystems of the satellite (power, thermal, data ...), its modes and other important concepts about a satellite mission, so that the simulator is as much approximated as possible to the scenario of a real mission.

Once the software improvements are implemented and validated a final satellite mission analysis example is elaborated. This mission consists in observing a geographical zone of the Earth with a real objective and constraints...

Finally, the result of the project is a GUIDE application from the Matlab software so that satellite mission analysis simulations can be programmed and studied from it. By analyzing the mission objectives and constraints, the user can try different satellite designs, orbits, modes, ground stations... and realize a comparison between them to obtain the best possible result to achieve their objectives.

The methodology used to develop the project is, firstly to develop a study of space mission typical parameters and concepts related to satellites from books and research papers so that a big theoretical background is achieved. With this knowledge acquisition, the existing software can be understand and improved by implementing them to the Matlab Software in the best possible way.

The validation of this project is done by comparing the results of every new implementation, by itself and on the whole, with the first version of the simulator and with the theoretical background studied to check its reliability.

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# INTRODUCTION, MOTIVATION AND RESEARCH OBJECTIVES

Nowadays the number of satellites in orbit performing space missions is increasing due to the useful applications and advantages that they can bring us such as enable communications or acquiring important data. A space mission has a very high difficulty as it is composed of lots of subsystems and all of them have to work properly and in the most efficient way during launch and under the space environment conditions. Furthermore, putting a satellite into orbit and the technology used to build it are very expensive.

Simulators help humans to establish a connection between reality and a computer environment. They generate specific and complex scenarios that are not easy to produce on Earth due to their difficulty and high cost. Therefore, in order to ensure safety and reduce the risk of the mission, tests and simulations must be performed and analyzed to improve and adjust as much as possible the satellite system.

The idea of this thesis is to perform and improve a Matlab software simulating an Earth Observation Mission with satellites, which takes into account the different main subsystems of the satellite and studies its behaviour in the space environment conditions in order to obtain a simulation as close as possible to a possible real mission.

The simulation software developed will be a very useful tool for companies to test the parameters of their own products and see the viability of the mission before performing any experiment or test and in this way, safe large quantities of money and time.

The principal subsystems of the satellite that are developed are the power system, thermal system, data budget, communication system, payload. Also, many different important concepts about a space mission and space environment must be studied and analyzed such as duty cycle, revisit time, access time, eclipses of the satellite depending on the position in its orbit, lifetime of the mission and the orbital propagator.

Every satellite equipment such as the batteries or solar panels, has been implemented with real data from GOMSpace available products in order to elaborate a more realistic mission analysis.

Firstly, a theoretical background study will be done in order to understand every part of a satellite space mission. Once the basic theory is learned, the first version of the simulator code will be described with examples. Then the software improvements on different parts of the software will be exposed.

The final idea of the thesis is to produce a GUIDE application with Matlab that contains every work developed. Thanks to this GUIDE, given a mission objective for Earth observation and a characteristic satellite, the user can try different satellite and orbit configurations in order to achieve them in the best effective way by comparing the results of the data provided from the software.

# CHAPTER 1. THEORETICAL BACKGROUND FOR A SATELLITE MISSION ANALYSIS FOR EARTH OBSERVATION

In order to develop and implement a satellite simulator, several theoretical concepts from the fields of science and engineering must be studied. Basic concepts such as coordinate systems or the possible orbit models to satisfy in a better way our objectives must be understood.

Another subject of study is the space environment which will affect the mission design so it is important to know the main sources that affect the satellite motion on orbit. The satellite itself is a complex machine composed by different subsystems, everyone with a different and vital function.

A deep study into all this subject must be developed in order to firstly understand the existing software and then apply improvements.

## 1.1.. Reference Coordinate Frames

### 1.1.1.. Earth-centered Inertial - ECI

This reference frame has its origin at the center of mass of the Earth. The z-axis coincides with the rotational axis of the Earth, the x-axis points towards the vernal equinox and y axis is computed by the right hand rule. X and y-axis coincide with Earth's equator. Inertial reference system so it is not fixed with respect to Earth's surface in its rotation.It is usually applied for orbital analysis of an Earth orbiting satellite.

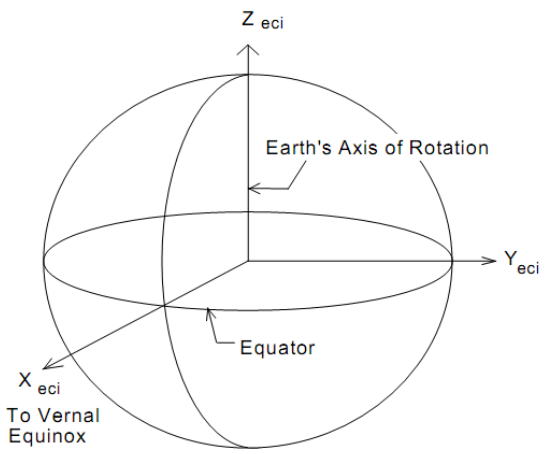



Figure 1.1: ECI Coordinate System representation

### 1.1.2.. Earth-centered Earth-fixed - ECEF

Origin at the center of mass of the Earth. Z-axis pointing to the North-Pole and does not coincide with Earth's axis of rotation. X-axis coincides with the 0$^o$ latitude and y-axis coincides with the equator. Both rotate with the Earth at $\omega_{Earth} = 7.2921 * 10^{-5} rad/s$. This system is useful to position astronomical objects such as planets.

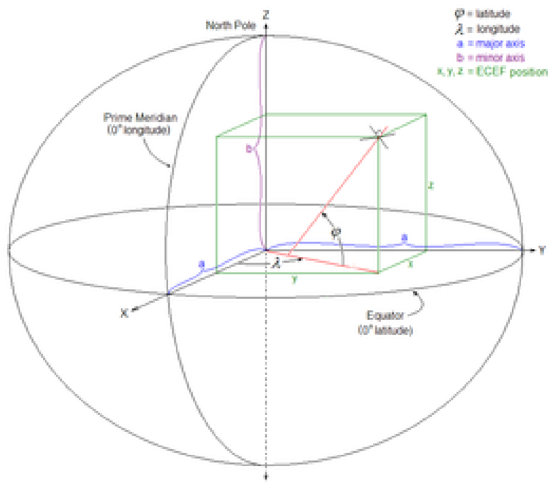



Figure 1.2: ECEF Coordinate System representation

### 1.1.3.. Geographic Coordinate System

Reference frame that allows to describe any geographic position on the on the surface of Earth using spherical measurements of latitude and longitude along with the elevation. It is a useful coordinate system to project the position of the satellite on the Earth surface.

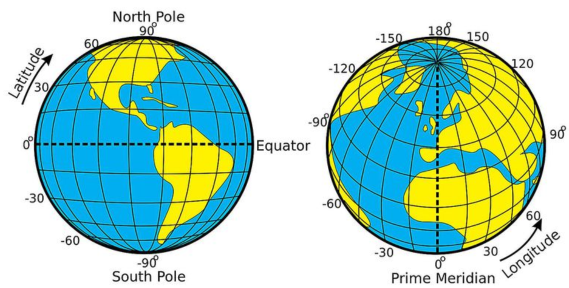



Figure 1.3: Geographic coordinates representation

### 1.1.4.. Orbit Frame

The origin is located at the center of mass of the satellite, with z-axis pointing towards Earth's center, perpendicular to the xy-plane. The x-axis pointing to flight direction and y-axis perpendicular. In a circular orbit, the entire frame rotates with $\omega_o rbitvelocity$, and the x-axis also coincides with the velocity vector.

### 1.1.5.. Body Frame

Local frame which the attitude measurements are made and originally described. Origin also located in the center of mass of the satellite, z-axis points the nadir and x and y axis are orthogonal to the rectangular sides of the cube.

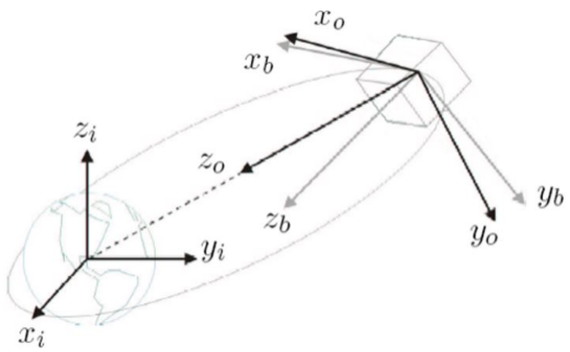



Figure 1.4: Body, orbit and ECI frames compared between them

### 1.1.6.. Euler Rotation Matrices

Euler transformation angles are used to transform from one coordinate system to another, or to rotate within one frame. In order to do this, three angles for three axis of rotation are needed: roll($\phi$)$, pitch(\theta) and yaw(\psi)$ for rotation around x, y and z axis respectively [2]:

$$R_x(\phi) = \begin{pmatrix} 1 & 0 & 0 \\ 0 & cos(\phi) & -sin(\phi) \\ 0 & sin(\phi) & cos(\phi) \end{pmatrix} \tag{1.1}$$

$$R_y(\theta) = \begin{pmatrix} cos(\theta) & 0 & sin(\theta) \\ 0 & 1 & 0 \\ -sin(\theta) & 0 & cos(\theta) \end{pmatrix} \tag{1.2}$$

$$R_z(\psi) = \begin{pmatrix} cos(\psi) & -sin(\psi) & 0 \\ sin(\psi) & cos(\phi) & 0 \\ 0 & 0 & 1 \end{pmatrix} \tag{1.3}$$

## 1.2.. Orbit Design

Depending on the mission and the characteristics of the satellite, the orbital parameters can be very different, so it is important to select a good design for the orbit to accomplish Earth observation missions.

### 1.2.1.. Orbit type

From the altitude point of view there are 3 types of orbit: LEO (Low Earth Orbit) up to 1200 km, MEO (Medium Earth Orbit) up to 35000 km and HEO (High Earth Orbit) above.

The best type for Earth observation is the LEO orbit. The satellite is positioned as close as possible to clearly see the surface in circular orbit. However, it also has disadvantages as the limited field of view to communicate or the drag force. Another advantage of LEO is that the cost of putting satellites into this orbits is reduced compared to others, so constellations can be placed easily.

### 1.2.2.. Orbital Parameters

The orbital elements allow to define a specific and unique orbit . There are many ways of describing an orbit, but the most commonly used in celestial mechanics are the Keplerian Elements:

- Eccentricity (e): shape of the orbit, being 0 circular, from 0 to 1 elliptic, 1 for parabolic and bigger than 1 for hyperbola.
- Semi-major axis (a): longest diameter of the shape of the orbit.
- Inclination (i): vertical tilt of the orbit with respect the equator. Coincides with the latitude angle.
- RAAN $(\Omega)$: the right ascension of the ascending node is the angle between the ascending node (point where the orbit crosses the equator from South to North) with respect to the reference frame's vernal point.
- Argument of periapsis (w): angle between the ascending node and the perigee of the orbit.
- True anomaly $(\nu)$: angle between the perigee and the position of the orbiting object at a specific time (epoch).

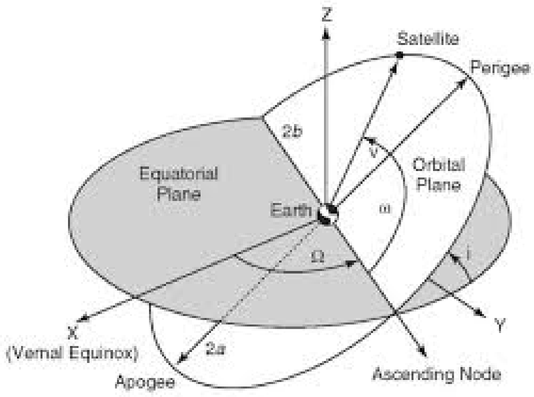



Figure 1.5: Orbital Keplerian elements

Another way of showing these orbital parameters are the Two-line Elements (TLE) [32]. It is a data format that encodes the Keplerian elements in addition with more information of the satellite, commonly used for modern computer software.

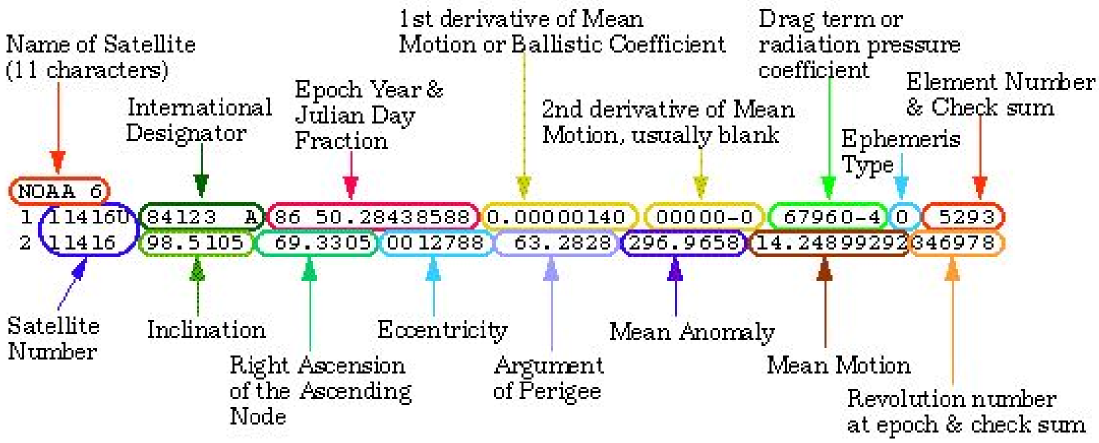



Figure 1.6: Definition of Two-line Element Set Coordinate System from NASA

## 1.2.3.. Orbit propagators

Keplerian elements define a specific orbit in a specific time named epoch. In order to propagate the satellite along time, different methods can be used. The ones used in the project are the following:

### *1.2.3.1.. Linear Orbit Prediction*

It is the most basic orbit elements set, assuming a perfectly spherical central body and zero perturbations which not very accurate. All orbital elements are constants except the mean anomaly which changes linearly with time, scaled by the mean motion:

$$n_{ma} = \sqrt{\frac{\mu}{a^3}} \qquad \text{[rad/s]} \quad (1.4)$$

where $\mu$ is the standard gravitational parameter for Earth:

$$\mu = G \times M \qquad [\mathrm{m}^3/\mathrm{s}^2] \quad (1.5)$$

So, if at any instant $t_o$ the orbital parameters are [ $e_o, a_o, i_0, \omega_0, \omega_0, Ma_0$ ], then the elements at time $t_o + \delta t$ is given by $[e_o, a_o, i_0, \omega_0, Ma_0 + n_{ma} \times \delta t]$.

#### *1.2.3.2.. Numerical Integration Prediction*

This method is more complex but more accurate. It can include perturbations such as non spherical Earth, Moon and planetary gravity effects or drag forces due to the atmosphere. All this perturbations constantly affect the orbit of the satellite so it is important to take them into account to achieve a precise simulation. Uses numerical integration methods to compute the ECEF or ECI satellite coordinates.

## 1.3.. Satellite Subsystems

Modern satellites are an extremely complicated piece of equipment composed lots of subsystems. They live and die in space under extreme conditions, so all the subsystems must be interconnected and balanced to ensure maximum safety and effectiveness. All of them depend on each other so a global design between them must be done.

The function of the principal satellite subsystems modeled in this software are explained.

### 1.3.1.. Power Budget

Being in outer space and developing a concrete mission requires an energy inputs. The more powerful, clean and convenient source of power for satellites in space is the Sun. This budget is in charge of converting the Sun's radiation (mainly light and UV) into electrical power to supply energy to the system. This is done with solar panels of semiconductor photo-voltaic cells. However, as the satellite is placed in near LEO orbits, eclipses are quite frequent.

Therefore, batteries as a second source of power must be available for this periods of lack of radiation. They are charged during non-eclipse periods and discharged during the eclipses or for an extra need of power. [4]

Another important equipment of this system is the Electronic Power Supply (EPS) [33]. The role of the EPS is to generate, store and distribute the electricity produced by the solar panels to all the satellite with a certain efficiency.

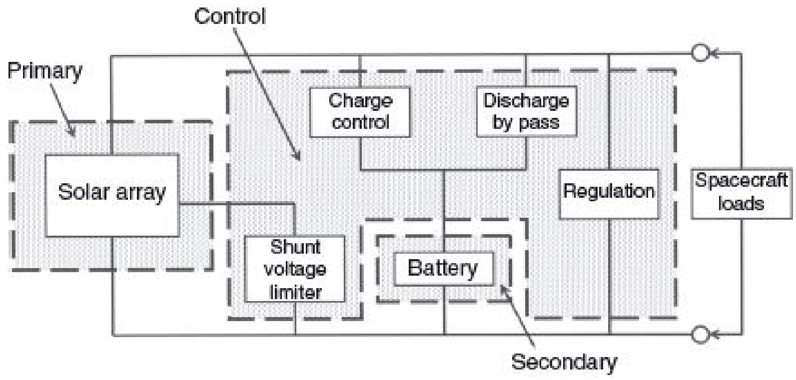



Figure 1.7: Schematic of typical spacecraft power system block elements

## 1.3.2.. Thermal Budget

The thermal system is in charge of controlling the level of temperature of the equipment and payloads. It is essential during all phases of a space mission to protect flight hardware and to guarantee the optimum performance and success of the mission.

In order to maintain the satellite temperatures within a set of parameters, a thermal control must be done. If the temperatures are out of the required ranges, the equipment could be damaged, its performance could be severely affected and the lifetime reduced. While being in space it would hardly be possible to correct such damages so that is why the thermal control systems need to be properly designed and tested to be highly reliable.

Lots of parameters allow us to design a correct thermal budget. The painting of the satellite produce emittance and absorptance ratios, which affect the balance of incoming and outgoing radiation. The incoming radiation is formed by the solar, albedo and planetary heat. Then the outgoing energy is composed by the heat that the satellite releases to the deep space which is at a temperature about -270$^o$C.
Another source of thermal energy is the dissipation of the components of the satellite by the Joule effect.[5]

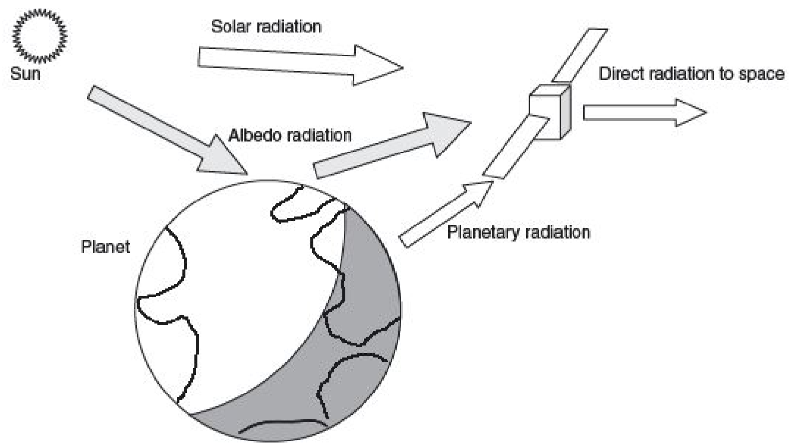



Figure 1.8: Typical spacecraft thermal environment

### 1.3.3.. Communication and Data Budget

The communication subsystem is composed by the satellite and the ground stations. In an up-link transmission, the ground station sends any command order to the satellite and in the down-link the satellite transmits the data collected for Earth observation, which is in fact the main goal of the mission. [6] To be able to establish communications, the main

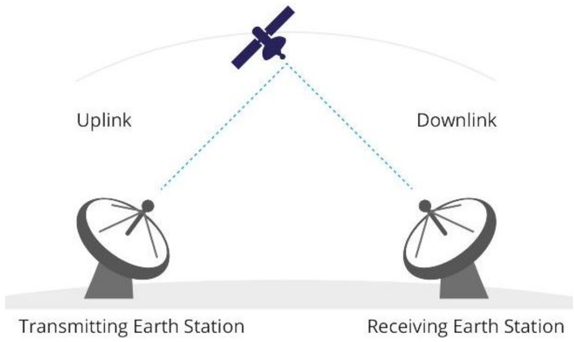



Figure 1.9: Representation of the up and down links between a satellite and the ground stations

constraint is to maintain a field of view between the satellite and the ground station. For LEO orbits, as the altitude is not very high, this field of view is restricted. This can be solved by positioning more than one satellite in specific points of the orbit, so a constellation is achieved.

When the down-link is not available, the satellite will keep taking photos for Earth observation, so they must be saved in the memory which has a limited capacity. Once the memory is full, the satellite can not obtain more data so the objective of the mission is not plenty achieved. That is why it is really important to design a good communication links so ground stations must be selected in the most optimal position.

### 1.3.4.. Attitude and Control Subsystem

Essential subsystem which continuously orientates the satellite in the optimal position and direction depending on the necessity and objective of the moment.

For Earth observation, the satellite payload must face the Earth at all times, pointing correctly to the desired geographic position, which is the attitude controller's responsibility. The controllers equipped on the satellite will also contribute to the power consumption. However, the satellite could have other priorities as projecting the solar panels to the Sun to gain power so the orientation changes constantly along the orbit.

Moreover, the perturbations of the space environment such as solar pressure or gravitational forces affect the positioning of the satellite so they must be correctly countered. Reference coordinate frames and Euler angle matrices are the basic way for the computation of the satellite orientation and positioning.

### 1.3.5.. Payload Subsystem

The payload of a satellite is the instrument carried to fulfill the mission final objective. A satellite can have multiple payloads for different types of operations in space. It is the main reason for a satellite to be deployed. All the other subsystems have to be designed around the payload so that the objective of the mission can be accomplished.

An typical example for Earth observation payload can be a high resolution camera to perform useful images for data processing.

## 1.4.. Key concepts of a space mission for Earth observation

To evaluate and address a concrete mission to observe a geographic zone of the Earth, some parameters must be studied and computed to specify the way the satellite must work to fulfill the objectives of a concrete mission.

### 1.4.1.. Revisit Period

The revisit time is an important consideration for a number of monitoring applications, especially when frequent imaging is required. From the point of view of the satellite, the revisit time is the elapsed time before the satellite retraces its path, passing over the same exact point on the ground surface. From the point of view of an Earth observation user, the revisit time is defined as the length of time to wait for the satellite system to be able to observe the same point on Earth. It is an important constraint for the design of space missions, very related to the type of orbit selected. To reduce the revisit period, constellations of LEO satellites can be used.

### 1.4.2.. Access time

The access time of a satellite is the measurement of the amount of time that the desired area of study can be observed. Depending on the orbit and therefore the velocity of the satellite the access time will vary. Also, the characteristics of the equipped camera such as the lent o will modify the parameter. As the revisit period, the access time is another mission constraint.

### 1.4.3.. Duty Cycle

The duty cycle is the fraction of one period in which a signal or system is active. Applied to the case of a space mission for Earth observation, the duty cycle of the satellite will be the fraction of time (usually minutes) per orbit in which the satellite is communicating with the ground station to download data. It also can be expressed in percentage by dividing the time with the orbit period.

### 1.4.4.. Swath and vertical swath

The swath of a satellite is the width of the area on the surface of the planet which is imaged by the sensor, which is the camera, during its movement along the orbit. The vertical swath is set as the length of the area. This parameter is useful to separate the area taken in continuous photos. [**?**]

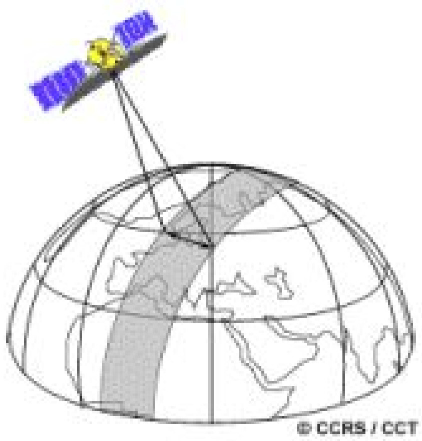



Figure 1.10: Representation of the satellite's swath on the Earth surface

### 1.4.5.. Overlap

Overlap is the percentage of the common area on consecutive images along the flight direction, in order to facilitate the assembly.

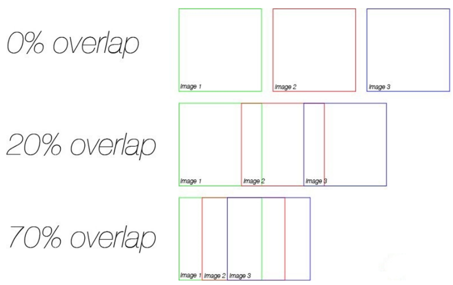



Figure 1.11: Example of different overlap configurations

# CHAPTER 2. SATELLITE MISSION ANALYSIS SIMULATOR FIRST VERSION

## 2.1.. Introduction to the existing software

Once a deep study of the theoretical background about the general concepts of a satellite space mission for Earth observation has been done, with all this acquired knowledge, the next step is to understand the existing code, which is the first version of the simulator. It is important to learn the structure of the code and its main functionalities to get used to it, leading to a viable future improvements and new implementations.

The simulator has been developed with MATLAB software. The core of the code is the main script, where everything is defined.

## 2.2.. Main Script

Firstly all the different constants used along the simulation for every part are defined. The first inputs of the code include the time duration of simulation and the more general constants that are commonly used during a space mission such as the Earth characteristics or gravitational and light constants. Another important parameter to take into account in a simulation software is the time between steps, which means time elapsed between samples taken. A high value of this parameter will make the simulation less accurate but faster. If a good and precise computation is required, the value should be around 1-10 seconds. Once this parameter is defined, the total number of samples can be obtained:

$$N_{step} = \frac{T_{simulation}}{T_{step}} \tag{2.1}$$

Then the constants introduced refer to the different parts of the satellite:

- Platform size
- Orbit characteristics
- Ground Stations localization
- Target Areas
- Camera characteristics
- Equipment: batteries, electric power supply, computer, solar panels, ADCS and transmitter.
- Material specifications
- Data flow

Afterwards, one by one, the different systems integrating the satellite are called from the main script. They are in separated functions which are developed in the next section.

# 2.3.. Subsystem Functions

## 2.3.1.. Payload Analysis

In this function the track dimensions are computed. With the orbital height h known, firstly the orbital period and the satellite's velocity is calculated with the following equations:

$$T = 2\pi\sqrt{\frac{\mu}{(h+R_E)^3}} \quad [s] \tag{2.2}$$

$$V_{sat} = \sqrt{\frac{\mu}{(h+R_E)^3}} \quad [m/s] \tag{2.3}$$

The ground track speed is the velocity of the satellite on Earth's surface and will vary depending on the orbit height:

$$V_{gt} = V_{sat} \times \frac{R_E}{h+R_E} \quad [m/s] \tag{2.4}$$

Once the ground speed is obtained, the visibility time of a concrete location can be calculated, which is an important parameter for the mission analysis for Earth observation.

$$T_{obs} = \frac{S_v}{V_{gt}} \quad [s] \tag{2.5}$$

Also, the camera characteristics are analyzed. However, they are not used during the project so it is not necessary to develop them. The payload is considered as a camera that takes pictures without taking into account its details.

## 2.3.2.. Orbital Analysis

The first and most basic subsystem of a satellite is the orbital propagator. It is the base for the other subsystems as everything depends on it. It is in charge of the computations of the variations of the satellite's position and velocity, which are computed in different coordinate systems to use them in the following subsystems. Also,there is the possibility to introduce a constellation of satellites up to 4.

In the theoretical background part, two types of propagators have been studied. In the existing simulator, the most simple propagator which is linear orbit prediction has been used to simulate the satellite's motion.

Firstly, the Keplerian elements are defined:

- Semi-major axis:

$$a = h + R_E \quad [km] \tag{2.6}$$

- Inclination: i defined by the user
- Eccentricity: set e = 0 for circular orbit

- RAAN: $\Omega_1$=0$^o$ (default value, it could be any)
  If there is more than one satellite (constellation), 90$^o$ of separation between each one: $\Omega_2 = \Omega_1 + 90^o, \Omega_3 = \Omega_1 + 180^o, \Omega_4 = \Omega_1 + 270^o$.
- True anomaly: for circular orbits, the true anomaly is undefined as the perigee does not exist: $T_{a_{1,2,3,4}} = 0$,
- Argument of periapsis: for the same reason, it is assumed that the $\omega isplacedat \Omega, so:$ $\omega_{1,2,3,4} = 0$

### 2.3.2.1.. RAAN analysis

The initial RAAN is set to 0, and once the simulation starts, it is just affected by Earth's sidereal rotation velocity which is defined with a value of $\omega_{sidereal} = 7.29 \times 10^{-5} rad/s$. Therefore, it completes a full rotation of 360$^o$ every sidereal day. To compute this, the angular step has been defined as:

$$Angular_{step} = \omega_{sidereal} \times T_{step} \qquad [rad] \quad (2.7)$$

Iterating for the simulation time desired:

$$\Omega = \Omega_o + Angular_{step} \times N_{step} \qquad [rad] \quad (2.8)$$

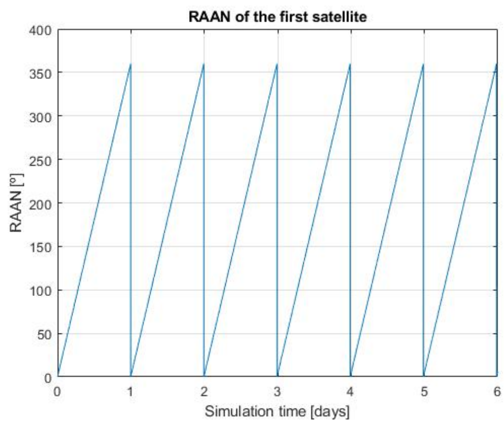



Figure 2.1: Representation of the RAAN evolution along 6 days

### 2.3.2.2.. True Anomaly analysis

The same procedure as for the iteration of the RAAN is done. However, the true anomaly depends on the orbit height so the angular step is obtained from the period, rotating once 360$^o$ for every period.

$$Angular_{step} = \frac{2\pi \times T_{step}}{T} \qquad [rad] \quad (2.9)$$

and so, the iteration:

$$T_a = T_{a_o} + Angular_{step} \times N_{step} \qquad [rad] \quad (2.10)$$

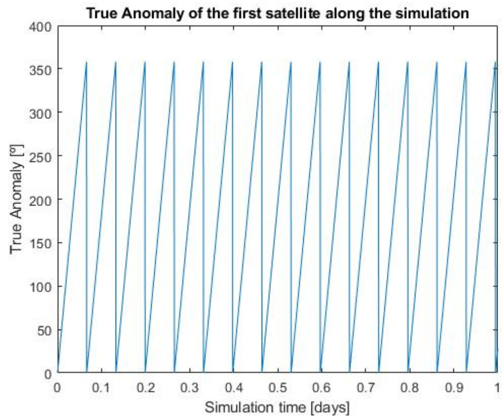



Figure 2.2: Representation of the true anomaly evolution during 1 day, for a period of T= 1.59 hours = 0.066 days

### *2.3.2.3.. Orbital Propagator*

As stated before, the orbital propagator uses linear variation. To do this, the mean anomaly must be computed. For circular orbits they have the same value as there is no periapsis, but when $e > 0$ the true anomaly referred to the real elliptical orbit defers from the mean anomaly of the hypothetical circular orbit.

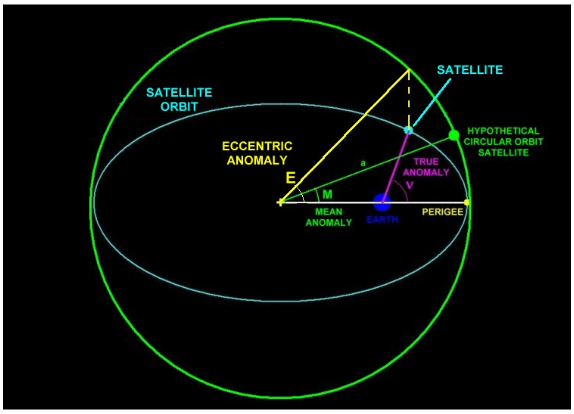



Figure 2.3: Schematic representation of the difference between $T_a$ and $M_a$

.

By trigonometry relations, the computation of the true anomaly is done:

The initial eccentric anomaly is obtained from the four quadrant inverse tangent:

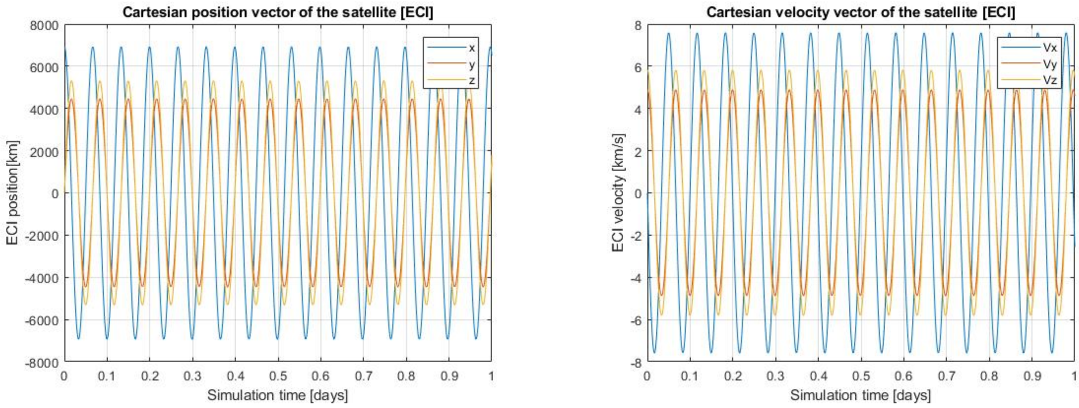
$$E_o = atan2\Big(\frac{\sin(T_{a_o}) \times (1-e_o)^{0.5}}{\cos(T_{a_o}) + e_o}\Big) \quad \text{[rad]} \quad (2.11)$$

Then the initial mean anomaly can be calculated:

$$M_a = E + e \times \sin(e) \quad \text{[rad]} \quad (2.12)$$

Once all the initial Kepler parameters are defined [ $e_o, a_o, i_0, \omega_0, \omega_0, M_{a_0}$ ], several open source MATLAB functions are implemented to transform this into ECI, ECEF and Geographic coordinates. Below, a one day simulation has been computed:

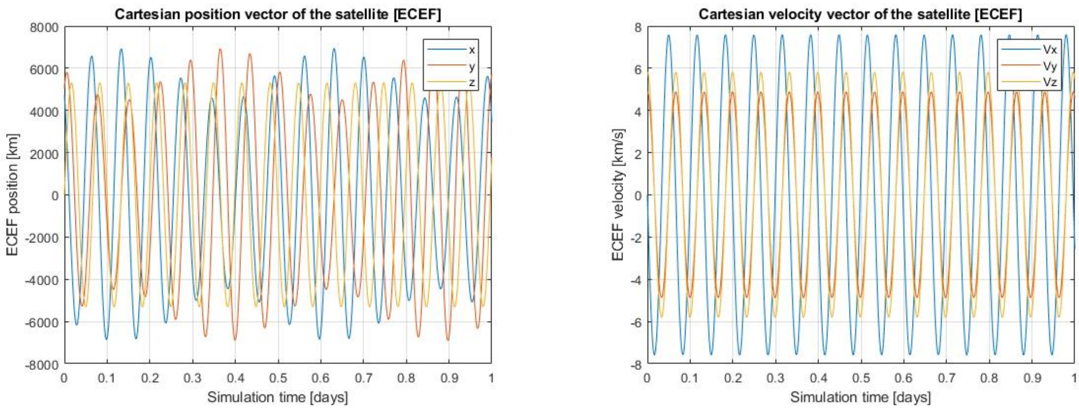
- ECI REFERENCE SYSTEM




(a) ECI position (b) ECI velocity

Figure 2.4: ECI computation

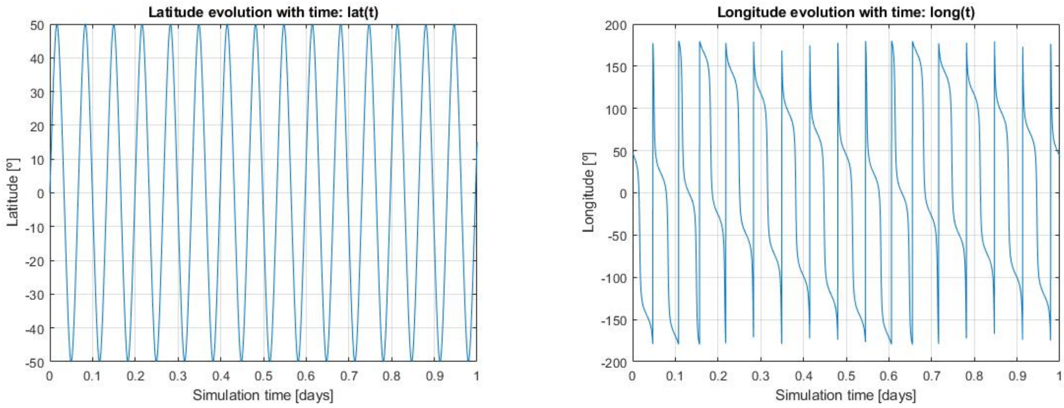
- ECEF REFERENCE SYSTEM




(a) ECEF position (b) ECEF velocity

Figure 2.5: ECEF computation

- GEOGRAPHICAL REFERENCE SYSTEM




(a) Latitude evolution

(b) Longitude evolution

Figure 2.6: Geographical computation

As expected, a linear variation orbital propagator produces a soft evolution of the coordinates. To check its reliability the results are analyzed. For both ECEF and ECI the module of the velocity is maintained constant along the simulation (around 7.6 km/h) as the altitude of the orbit does not change as its circular and there is lack of perturbations . Velocity only changes its vectors direction due to the inclination of the orbit which in this example is of i=50$^{o}$. Longitudes go from 0 to 180$^{o}$ or -180$^{o}$ every time an orbit is completed corresponding to a period of time T, while latitudes achieve maximum and minimum values around the inclination degree.

### 2.3.3.. Global coverage

This part of the simulator gives essential information in order to obtain a good mission design. Once the coordinates of the satellite are computed during all the simulation thanks to the orbital propagator an analysis of the track must be done.

To do this, an Earth map is computed where, every time the satellite passes nearby a concrete position, one step in the altitude is included. For this tool, geographical coordinates (latitude and longitude) have been used as the input of the function. So the zones where the satellite has been more times will have a higher altitude that the zones where it does not arrive. This will give us useful information to change the orbit design so that we obtain higher visualizations of our zone for Earth observation.

With the same inputs of orbit as the example before, the following map has been generated:

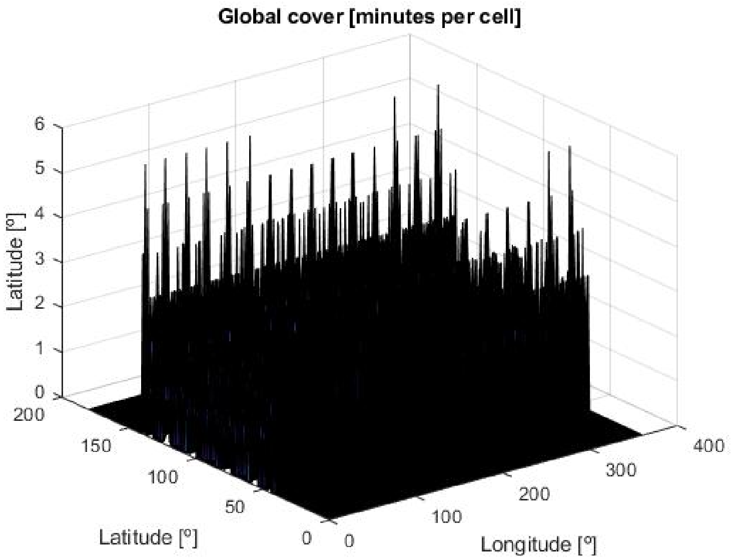



Figure 2.7: Global coverage for 7 day simulation for an orbit of i=50º and h=550km with one satellite

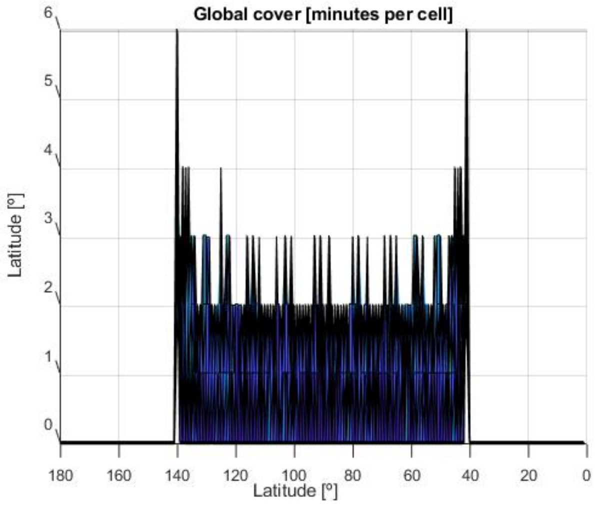



Figure 2.8: Lateral view of global coverage for 7 day simulation for an orbit of i=50º and h=550km with one satellite

It can be clearly seen that main orbit parameter to take into account is the inclination. It must be near the latitude values of the zone that wants to be observed because the satellite will cover it more times which is commonly desired. In the case shown, the satellite covers the latitudes of 50º 8 times while other zones are just covered twice.

As the time simulated increases, the difference between most and less seen zones will increase so it is important to do a good orbit design.

### 2.3.4.. Ground Station, Target Areas and Ground track

#### *2.3.4.1.. Ground Station configuration*

While being in orbit, the satellite accumulates the data taken (images for example) in its memory, which has a limit. If the memory reaches it, the satellite won't be able to take more images so the mission fails. To be able to collect as most data as possible a down link must be available. In order to make this possible the ground stations location must be well placed in strategic geographical zones, such that along the mission the full memory state is not achieved in a way that data is not lost.

An approximation to obtain when the down-link will be available is done with the field of view. While the satellite is in field of view with the ground station the download of data can happen.

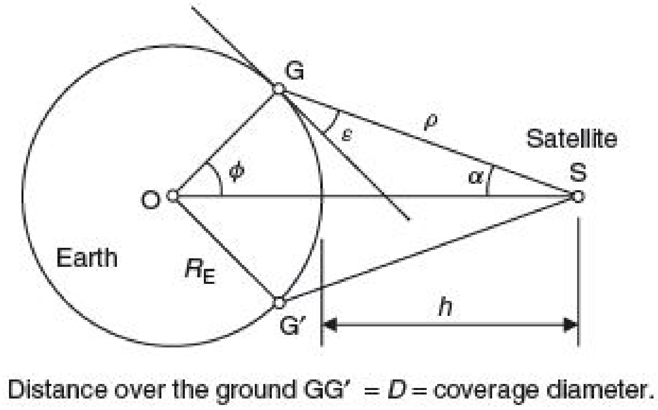



Figure 2.9: Representation of the field of view of a satellite

where:

- G ground station position
- $\alpha$ nadir angle
- $\rho$ range
- h orbit height
- $\varepsilon$ GS elevation
- $\phi_E$ center angle

The objective of the FOV analysis is to determine the maximum range $\rho$ at which the satellite can download data. Firstly the center angle is computed by trigonometry:

$$\phi_E = \frac{R_E \times \sin(90+\varepsilon)}{R_E + h} \tag{2.13}$$

Then the nadir angle must be:

$$\begin{aligned} 90 &= \alpha + \varepsilon + \phi \\ \alpha &= 90 - \varepsilon - \phi \end{aligned} \tag{2.14}$$

Finally, the maximum range is:

$$\rho = R_E \times \frac{\sin(\alpha)}{\sin(\phi)} \tag{2.15}$$

As it can be observed, the maximum range will only depend on the altitude and the elevation of the ground station.

For LEO orbits, the altitude is not very high so this implies a limited FOV, so a constellation may be needed or several number of GS activated to download data.

About the elevation constraint, an important feature of the ground station characteristics is its horizon mask, which defines the region of the sky within which the ground station can communicate with the spacecraft. Obstacles such as surrounding mountains, buildings and other antennas will determine the minimum elevation of the antenna. [6]

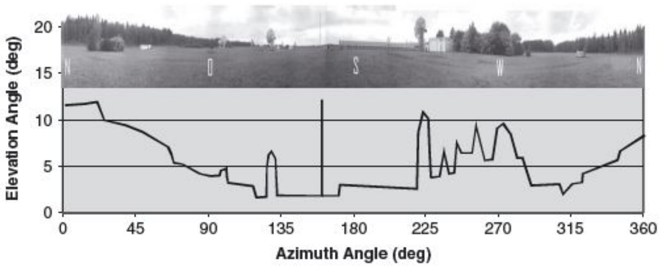



Figure 2.10: An example of a horizon mask, illustrating the constraints on minimum elevation on a ground station

So, depending on the surroundings of the ground station used and the orbit high designed, the communication availability between the satellite and the GS will vary.

Finally, during all the simulation time, for each iteration the distance from the satellite to the ground station will be computed:

$$\rho_{real} = \sqrt{(X_{sat} - X_{gs})^2 + (Y_{sat} - Y_{gs})^2 + (Z_{sat} - Z_{gs})^2} \tag{2.16}$$

And if:

$$\rho > \rho_{real} \tag{2.17}$$

the communications will be activated so that if the satellite has data in the memory, it will be downloaded.

### *2.3.4.2.. Target Areas*

The target areas are the Earth locations where the satellite must pass by and collect the data, to fulfill the mission for Earth observation goal. They are defined as a latitude, longitude and altitude respect sea level of the center of the location, plus a radius to cover the zone.

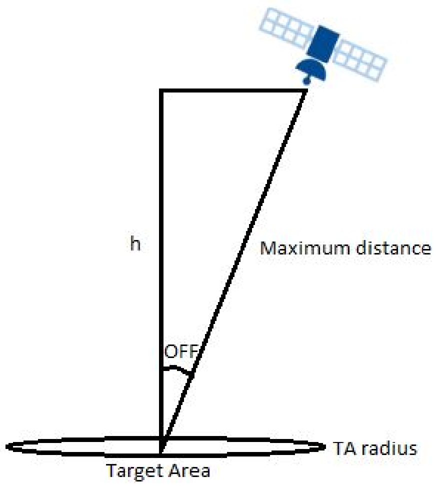



Figure 2.11: Schematic of the target area distance configuration

The maximum range to enable to observe a desired target area is:

$$d_{max} = \sqrt{h^2 + (\tan(OFF) \times h + TA_{radius})^2} \tag{2.18}$$

Again, the same comparison procedure as before is done. If the distance between the satellite and the target area is less than the maximum range $d_{max}$ computed, the satellite will be able to acquire data from the target area.

To show the results, a simulation is done. For the same scenario as in the previous function shown, the results obtained for 7 days-duration are:

Being two target areas defined:

- Latitude 50$^o$
- Longitude 50$^o$
- Radius 3350 m
- Altitude 0 m

- Latitude 80$^o$
- Longitude 30$^o$
- Radius 3000m
- Altitude 0 m

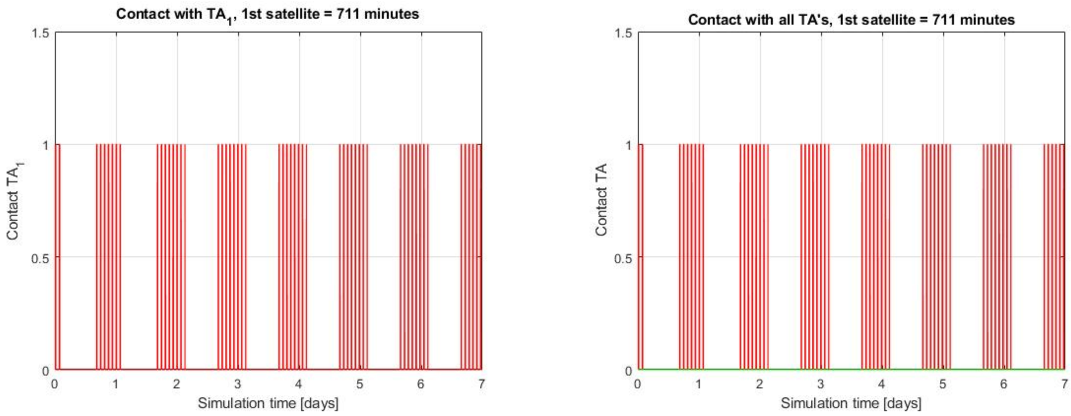
and two ground stations located at:

- Latitude 40$^o$
- Longitude -4$^o$
- Elevation 10$^o$
- Altitude 100 m

- Latitude 70$^o$
- Longitude 30$^o$
- Elevation 10$^o$
- Altitude 100 m

The following results are obtained:

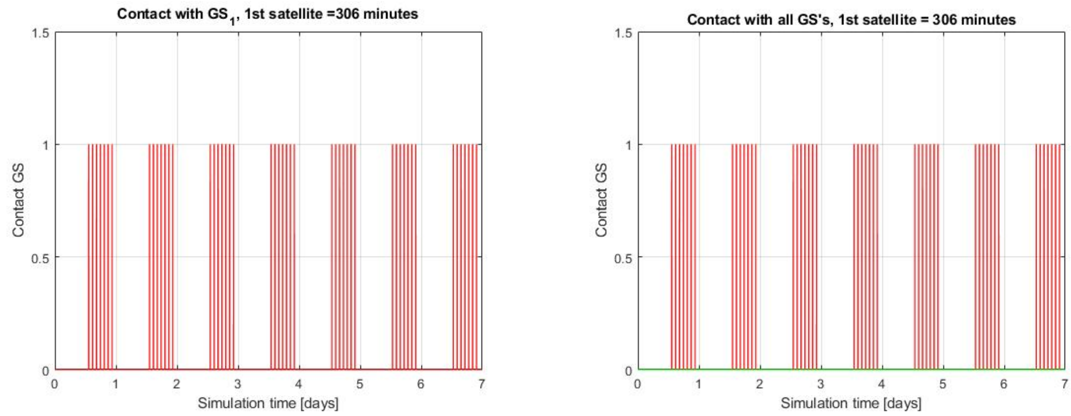
- Target Areas visualized




(a) Target Area 1 (b) All target areas

Figure 2.12: Target areas visualized

- Ground Station contacts




(a) Target Area 1 (b) All target areas

Figure 2.13: Ground Station contacts

When a contact is produced, a 1 value is plotted, while 0 are when there is no contacts. As the latitude of the second target area and ground station is much higher than the inclination of the orbit simulated, 0 contacts and 0 visualizations are produced, so the total of them are just the ones with the ground station 1 and target area 1.

Another thing to recall, is that only when a target visualization is produced and data is collected (conditions of this explained in the next functions), the ground station contact is enabled when there is field of view. If the memory is empty, no contact is produced.

### *2.3.4.3.. Ground Track*

A ground track is the path on the surface of a planet directly below an aircraft or satellite. In the case of a satellite, it is the projection of the satellite's orbit onto the surface of the planet, which in our case is the Earth.

A satellite ground track may be thought of as a path along the Earth's surface which traces the movement of an imaginary line between the satellite and the center of the Earth. In other words, the ground track is the set of points at which the satellite will pass directly overhead, or cross the zenith, in the frame of reference of a ground observer. This part of the function generates a ground track using an open-source code. The latitude and longitude of the satellite during a simulation are plotted on an Earth ground map, from -90 to +90 of latitude representing the y-axis and from -180 to +180 representing the longitude on the x-axis. The result for the same simulation as before is shown:

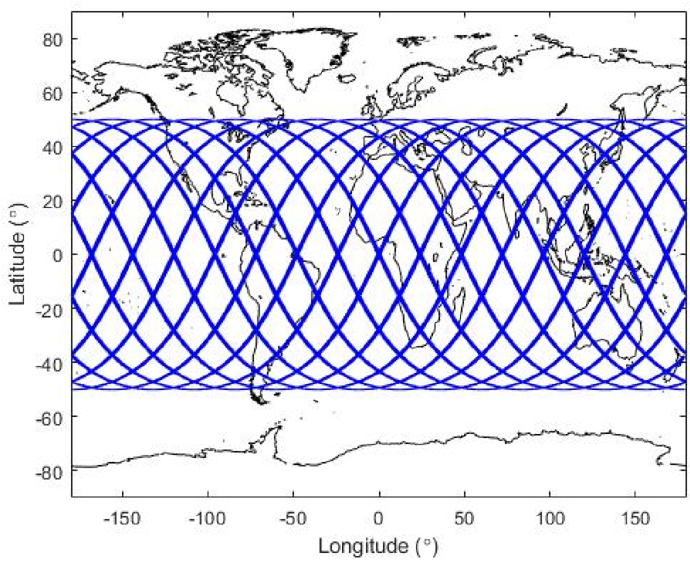



Figure 2.14: Ground track for a satellite on orbit of i = 50$^{o}$, h = 550 km during 1 day

## 2.3.5.. Solar Analysis

The Sun is an essential part of any satellite mission. To work properly, the satellite needs a source of energy, which will come mainly from the Sun's radiation. Also, part of this radiation will heat the satellite, which is also important to maintain an adequate thermal balance so that the hardware is not compromised.

### *2.3.5.1.. Sun Propagator*

The Sun's position $([X_{sun}, Y_{Sun}, Z_{Sun})$ and velocity $(V_{xSun}, V_{ySun}, V_{zSun})$ is computed respect to the Earth in ECEF coordinates with an open source function, which algorithm is based on a numerical approximation of the exact equations. This function gives the Sun zenith and azimuth angles from the observer introduced position in geographic coordinates.

To compute the Sun angles respect to the Earth, the latitude, longitude and altitude introduced are $0^{o}, 0^{o} and -R_E$ respectively. As the satellite coordinates are used in ECEF, a conversion function is used to pass this Sun azimuth and zenith values to ECEF coordinates.

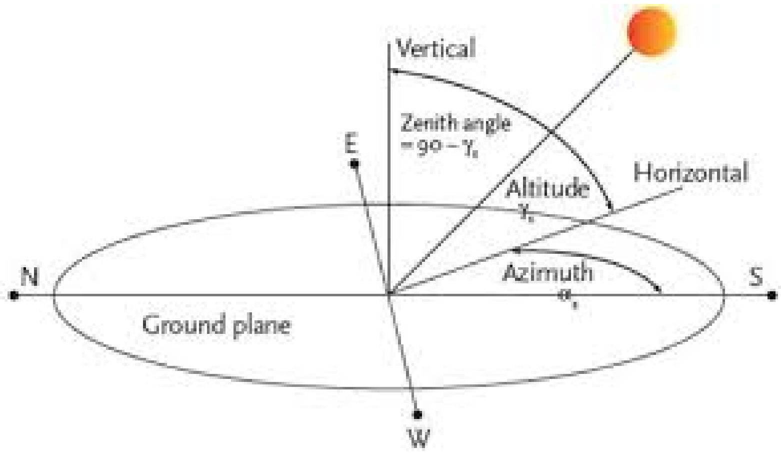



Figure 2.15: Representation of the zenith and azimuth of the Sun respect an observer position

### *2.3.5.2.. Solar projection on the satellite*

Once the position of the sun is obtained, an analysis between it and the satellite must be done. The satellite of this first version is a 6U small satellite, with the following area's values for the 6 cube faces:

- $Area_1 = Area_6 = 0.02m^2$
- $Area_2 = Area_5 = 0.06m^2$
- $Area_3 = Area_4 = 0.03m^2$

The projected area is the amount of area that the sunlight illuminates. Depending on the satellite's orientation respect to the Sun, the real radiation that arrives to its faces will change.

Firstly the normal vectors of each face and sun normal vectors should be computed:

- Area 1 will face the Earth center so:

$$\begin{aligned} N_{A1} &= [-X_{sat}, -Y_{sat}, -Z_{sat}] \\ N_{A6} &= [X_{sat}, Y_{sat}, Z_{sat}] \end{aligned} \tag{2.19}$$

- Area 2 will face towards the satellite's velocity:

$$\begin{aligned} N_{A2} &= [Vx_{sat}, Vy_{sat}, Vz_{sat}] \\ N_{A5} &= [-Vx_{sat}, -Vy_{sat}, -Vz_{sat}] \end{aligned} \tag{2.20}$$

- Area 3 normal vector therefore, is the perpendicular of the previous both vectors. So cross vector product is applied:

$$\begin{aligned} N_{A3} &= N_{A1} x N_{A2} \\ N_{A4} &= N_{A2} x N_{A1} \end{aligned} \tag{2.21}$$

- Sun vectors are computed for each face so that

$$N_{Sun_i} = [X_{sun} - N_{Ai_x}, Y_{sun} - N_{Ai_y}, Z_{sun} - N_{Ai_z}] \tag{2.22}$$

  being i the number of the face.

Then all this vectors are turned into unitary vectors by dividing its components with its module:

$$\hat{N} = \frac{[N_x, N_y, N_z]}{|N|} \tag{2.23}$$

Finally, to calculate the projection of the Sun on each area, the angle between both vectors $\alpha$is computed:

$$cos(\alpha_i) = \hat{N}_{A_i} \times \hat{N}_{Sun_i} \tag{2.24}$$

So the projected area $A_p$ is:

$$A_{p_i} = A_i \times cos(\alpha_i) \qquad [\mathrm{m}^2] \tag{2.25}$$

If the normal vector of a face points directly to the Sun, which means that it is parallel to the Sun's normal vector, the projected area will be the same as the area. As consequence, if both normal vectors are perpendicular, the area will be completely in the dark, so heat or energy won't be absorbed by it.

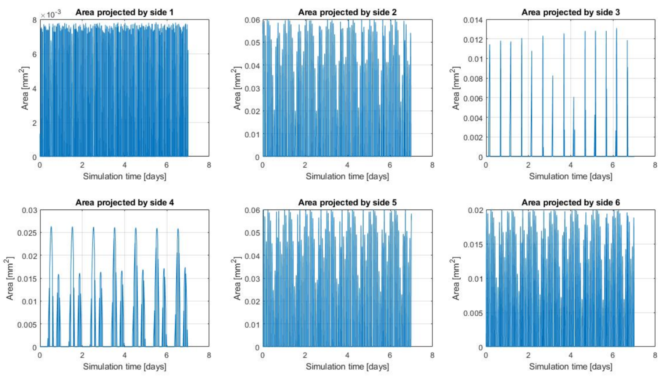



Figure 2.16: Projection of the satellite areas to solar radiation

As expected, the face 1 which points to the Earth to collect data receives almost any energy. Faces 2 and 5 are the ones which receive more energy so it is where the solar panels could be placed preferentially.

### *2.3.5.3.. Albedo*

The albedo is the percentage measure of the radiation reflected by a surface of the total received. Therefore, not only the solar radiation arrives to the satellite but also the reflected from other bodies. In this simulator, the Earth's albedo is taken into account which value can be approximated around 30%. However, depending on the geographical zone of the Earth, this value might differ. So an open source function [40] is used to obtain a more accurate albedo percentage depending on the geographical position of the satellite [36].

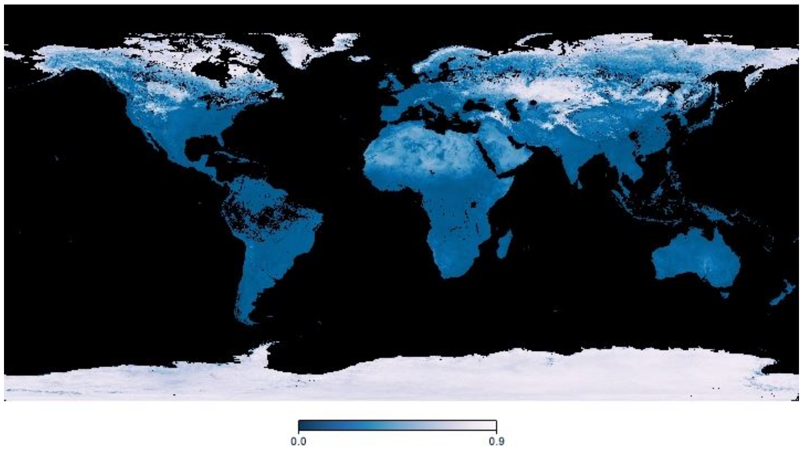



Figure 2.17: NASA's albedo measurement for February, 2017

Every calculation done before, is valid when the satellite is in the field of view of the Sun, but that does not happen every time. When the Earth is situated between the satellite and the Sun, the radiation received is only the corresponding to the albedo. So with the coordinates of the three involved bodies, the eclipse times are computed, obtaining an eclipse mask $Eclipse_mask = 0$ during eclipse time or $Eclipse_mask = 1$when no eclipse occurs.

For the same simulation conditions as before, the eclipse times obtained are:

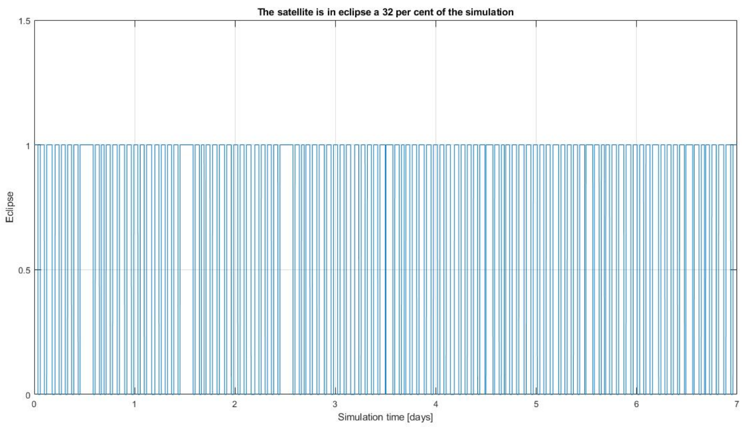



Figure 2.18: Eclipse computation time of 2.24 days during a 7 day simulation for an orbit at h=550 km

As higher the altitude of the orbit is, the number of eclipses will be reduced. However, its duration will increase. As the simulated is a LEO orbit, eclipses usually occur but with short duration.

### 2.3.6.. Power Subsystem

The power system consists to compute the balance between the input received power and the consumed power by the equipment. Satellite's energy will depend on its solar panels to receive power and on its batteries, that will be charged by them and will feed the different subsystems.

The first constraint set is that the voltage needed for any of the equipment should not be bigger than the batteries voltage, otherwise they could not be fed.

Then, the incoming power from the Sun and its albedo is computed. This incoming energy is absorbed by the solar panels placed on the surfaces of the satellite. The amount of $W/m^2$ that arrives to the Earth (distance of 1 AU) is called solar irradiation. It is approximated to a value of $S = 1400W/m^2$. Also, the efficiency of the solar panels and the EPS must be taken into account, as well as the area ratio between the complete face and the solar panels total area:

$$P_{Sun} = \eta_{panel} \times A_{ef} \times \eta_{EPS} \times A_p \times n_{panels} \frac{A_{panel}}{A_{face}} \times S_{sun} \times Eclipse_{mask} \quad [\text{W}] \quad (2.26)$$

This quantity of power S will be obtained also from the reflection by the Earth's albedo:

$$P_{albedo} = \alpha_{albedo} \times Eclipse_{mask} \quad [\text{W}] \quad (2.27)$$

Being the total incoming energy the sum of both:

$$P_{in} = P_{Sun} + P_{albedo} \quad [\text{W}] \quad (2.28)$$

This $P_{in}$ calculation is applied to each face which have different configurations. This power is the final absorbed by the panels, that for the simulation example, gives us the following values:

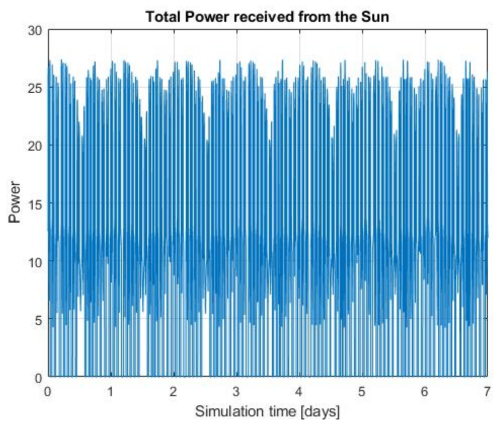



Figure 2.19: Power absorbed by the Sun and Earth's albedo

All this energy is used to feed the batteries. The capacity charge, at a time i, is computed by the following equation:

$$I_i = \frac{P_{in_i}}{V_{bat}} \times \frac{1}{3600} \quad [\text{Ah}] \quad (2.29)$$

If the charges on every step are added the following result for the previous simulation scenario is obtained:

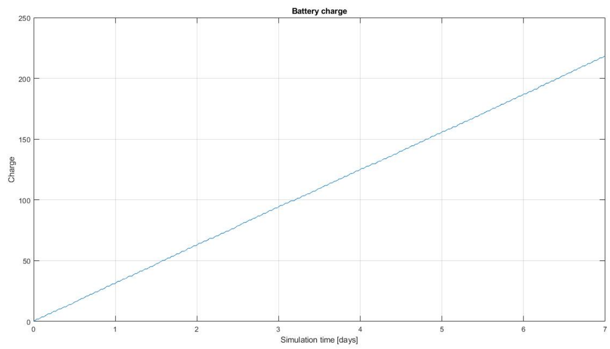



Figure 2.20: Battery charge for 7 days

Once the power incomes have been computed, the consumption is calculated. For every equipment that consumes energy, its consumption is calculated.

$$\Delta_{equipment} = \frac{I_{equipment}}{3600} \quad [\text{Ah}] \quad (2.30)$$

Every time that an equipment is activated, the consumption absorbs battery charge.

To make an analysis of the batteries lifetime, a DOD model is analyzed. DOD meaning Depth of discharge, is the percentage of the capacity that the battery has at every moment. Each kind of battery has a number of cycles depending the DOD percentage, which is given by the manufacturer. If this number is overcame, the battery dies and so does the satellite's mission. The battery implemented in the simulator gives us the following number of cycles:

- $N_{25\%} = 700$ cycles
- $N_{75\%} = 1400$ cycles

Meaning that if the battery goes under 25% of its capacity 700 times or under75% 1400 times, the battery dies.

The different DOD levels are computed with the following equation:

$$DOD_n = C_{bat} \times \frac{n}{100} \quad [\text{Ah}] \quad (2.31)$$

So, if the capacity goes under this $DOD_n$levels a cycle count is added.

To prevent this, a maximum discharge of the battery $DOD_{max} = 0.78$ (example of value) is introduced, so that if the battery reaches a certain percentage, the satellite stops to consume with the non-vital equipment until the battery is charged again.

In the following image, the result of this analysis is shown. In this simulation example, the battery remains almost full charged, so 0 cycles have been completed. Green, blue and red lines represent the $DOD_{90}$, $DOD_{75}$ and $DOD_{25}$ capacity values respectively.

## 2.3.7.. Thermal Subsystem

The thermal subsystem balance the heat absorbed and the heat emitted by the satellite, in order to obtain the satellite temperature. The main incoming heat sources are the Sun's

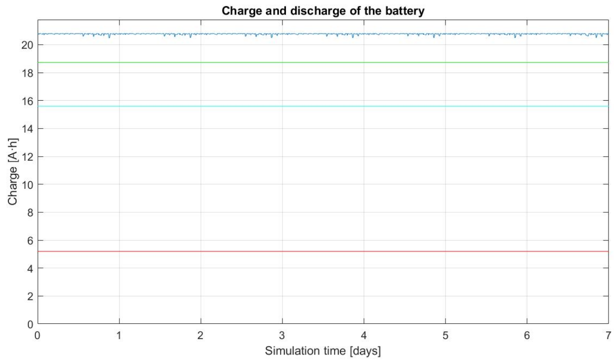



Figure 2.21: Battery DOD capacity analysis

radiation, Earth's albedo and Earth's radiation. This three are computed below:

Solar heat:

$$h_{Sun} = \frac{S \times A_p}{A_T} \times [\alpha_{panel} \times A_{panel} + \alpha_{chassis} \times (A_T - A_{panel})] \times Eclipse_{mask} \text{[W]} \quad (2.32)$$

where $A_T$ refers to the total area

$$A_T = A_1 + A_2 + A_3 + A_4 + A_5 + A_6 \qquad [\text{m}^2] \quad (2.33)$$

and $\alpha_{chassis}$ and $\alpha_{panel}$ refer to the absortance ratios of the panels and chassis

Albedo heat:

$$h_{albedo} = \alpha_{Earth} \times S \times [\alpha_{panel} \times A_{panel} + \alpha_{chassis} \times (A_T - A_{panel})] \text{ [W]} \quad (2.34)$$

Earth's heat:

$$h_{Sun} = E \times [\alpha_{panel} \times A_{panel} + \alpha_{chassis} \times (A_T - A_{panel})] \qquad \text{[W]} \quad (2.35)$$

where E is the Earth's radiation modeled as a black body at 300 k, with a value of $250W/m^2$

Another heat incoming source is the one produced by Joule's effect, that is the heat dissipated by the equipment used:

$$h_{Joule} = \eta_{equipment} \times \frac{V_{equipment}}{I_{equipment}} \qquad \text{[W]} \quad (2.36)$$

The vacuum of the space is considered to be at -270°C. A constant heat emission from the satellite to the vacuum is considered:

$$h_{out} = \sigma \times [\varepsilon_{panels} \times A_T + \varepsilon_{chassis} \times (A_T - Apanels) \qquad [\text{W/k}^4] \quad (2.37)$$

Once the heat gains and loses are obtained, the temperature balance can be calculated:

$$T_i = T_{i-1} + h_{Sun} + h_{albedo} + h_{Earth} + h_{Joule} - h_{out} \times \frac{T_{i-1} \times t_{step}}{m_{sat} * c_{sat}} \quad \text{[k]} \quad (2.38)$$

where $c = 897J/kgk$ is the heat capacity of the aluminum.

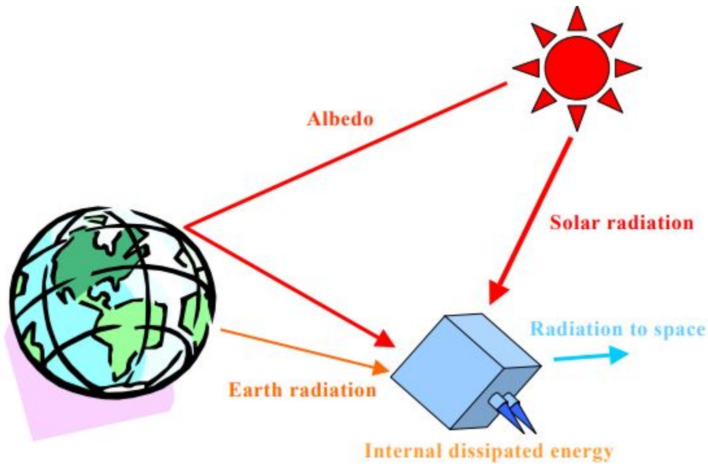
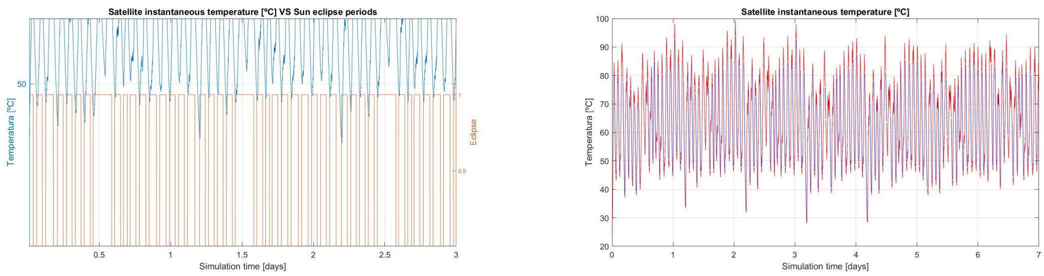



Figure 2.22: Representation of the thermal balance computed in the satellite

For the same simulation example as in the other functions, the results obtained are shown below:




(a) Temperature vs eclipse

(b) Satellite's temperature

Figure 2.23: Satellite's temperature in [k]

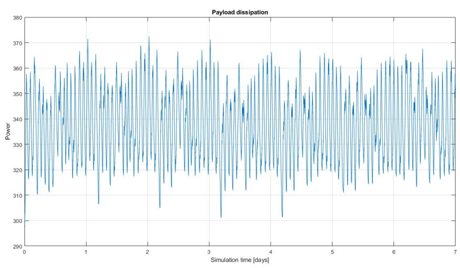



Figure 2.24: Satellite's power dissipation in Watts

In the first figure it can be observed how the temperature of the satellite falls during eclipses as the heat from the Sun which is the main source is eliminated. Also, the power dissipation varies in function of the equipment that is used in every step.

### 2.3.8.. Data Budget

Data subsystem consists in balancing the memory of the satellite. The memory is charged when the satellite passes by a target area range:

$$Data_{in} = TA_{mask} \times Camera_{charge} \qquad [\text{Mb}] \qquad (2.39)$$

where the camera charge is the Mb acquired by getting an image, which depends on the camera characteristics such as pixels and digitization.

The memory is discharged when the satellite passes by the range of an available ground station so that the down-link is possible to download the data collected:

$$Data_{out} = GS_{mask} \times V_{download} \qquad [\text{Mb/s}] \qquad (2.40)$$

So, the memory budget is:

$$Memory = Data_{in} - Data_{out} \times t_{step} \qquad [\text{Mb}] \qquad (2.41)$$

In the following plots, there is an example of a simulation result:

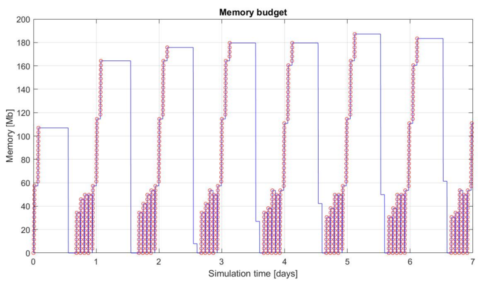



Figure 2.25: Memory budget balance for a 7 day simulation

### 2.3.9.. Scheduler

The functionality scheduler of a satellite is to relate the mapping tasks like observations, communications, down-links or control maneuvers to the satellite resource's. Satellites orbiting in space are complex systems, with lots of constraints imposed by the environment. That is why the scheduler must optimize all this constraints in order to fulfill the mission goal in the best way. When it is not possible to satisfy all requests for satellite operations, a priority system is typically used to help choose which tasks to schedule. The scheduler must also guarantee the safety of the equipment and its durability, so that the satellite life is not compromised.

The following scheduler constraints are used:

Collecting data$(Camera_{mask} = 1)$ when:

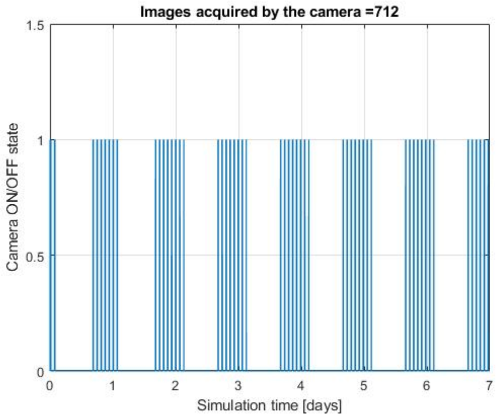



Figure 2.26: Photos acquired of the TA for a 7 day simulation (1 photo taken, 0 not)

- $T_{max} > T_{sat} > T_{min}$ temperature between the accepted ranges
- $P_{available} > Pinitial \times DODmax$ power available not under the maximum discharge allowed
- $Memory < Memory_{max}$ memory not full
- $TA_{mask} = 1$ satellite in the target area range
- $Eclipse_{mask} = 1$ no eclipse

Downlink communication $Downlink_{mask}$ when :

- $T_{max} > T_{sat} > T_{min}$ satellite's temperature between the accepted ranges
- $P_{available} > P_{initial} \times DOD_{max}$ power available not under the maximum discharge allowed
- $GS_{mask} = 1$ satellite in the ground station range

Attitude Determination and Control System can be activated when:

- $T_{max} > T_{sat} > T_{min}$ satellite's temperature between the accepted ranges
- $P_{available} > P_{initial} \times DOD_{max}$ power available not under the maximum discharge allowed
- $Eclipse_{mask} = 1$ no eclipse

Heater switched on when:

- $T_{sat} < T_{heater}$ satellite's temperature under the temperature set by the user to activate the heater

Primary on board computer consumes when:

- $T_{max} > T_{sat} > T_{min}$ satellite's temperature between the accepted ranges
- $P_{available} > P_{initial} \times DOD_{max}$ power available not under the maximum discharge allowed
- $Downlink_{mask}$ or $Camera_{mask}$ are activated

Secondary on board computer consumes when

- $Downlink_{mask}$ is activated

# CHAPTER 3. SOFTWARE IMPLEMENTATIONS

Once the theoretical background is studied and the first version of the simulator has been completely analyzed and understood, with this knowledge acquisition a deep research using reliable sources is done to define possible improvements that can be implemented to the software's first version.

Along this chapter, a development of this improvements is done. The goal of the project is to make a satellite mission analysis software as real as possible in order to achieve better simulations and so, realize a better mission analysis for future applications, and this will be possible by applying numerous code improvements.

## 3.1.. High Precision Orbital Propagator

As it has been seen before, the simulator uses a linear variation propagator, that only simulates the satellite's position by initial Keplerian elements and the Earth. No perturbation effects are considered, so it is not a very precise approximation to the reality.

An open source precision propagator [42] has been adapted and implemented for satellites for Earth observation for this software. This kind of propagator works by numerical integration prediction, so it is a more complex code, requiring more computation time, but more accurate. Concretely it uses a Radau second order integrator. It includes several perturbations and real planetary ephemerides. In order to do a correct code implementation of it to the existing software, several parts have been changed and adapted. The propagator consists of:

### 3.1.1.. Real data acquisition

#### *3.1.1.1.. Earth Orientation Parameters - EOP*

EOP are a collection of parameters that describe irregularities in the rotation of the Earth.

The Earth's rotational velocity considered in the first version $\omega_{Earth} = 7.2921 \times 10^{-5} rad/s$ is not constant over time. Any motion of mass in or on the Earth causes a slowdown or speedup of this $\omega_{Earth}$, or a change of rotation axis. Small motions are non-measurable, but movements involving large masses, like sea currents or tides, can produce detectable changes in the rotation. Technically, this EOP provide the rotational transform between ECI and ECEF as function of time.

This data has been changed, in order to make an update. From CelesTrak [38] the EOP data has been obtained from 2014 to 2019. If future dates are simulated, as they only provide 180 predicted days, the code has been changed so that values from the last year obtained are used as an approximation.

The main irregularities are precession and nutation phenomena [44]:

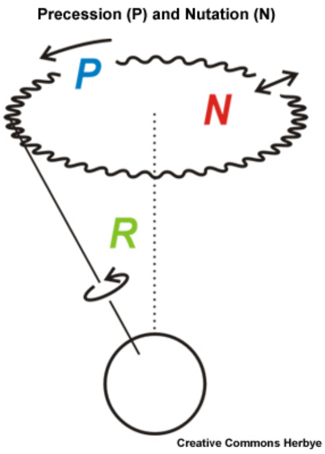



Figure 3.1: Representation of the precession, nutation and rotation of Earth's axis

#### *3.1.1.2.. Solar Indices and Geomagnetic Storm*

Solar indices are a series of parameters that describe the solar irradiation activity through the time while the geomagnetic storm values describe the disturbance of Earth's magnetic field due to solar winds. This parameters directly affect the atmosphere density which characteristics are needed for the drag calculation of the satellite. Data has been updated from Space Environment Technologies web page [38] until the 2019, so again if future dates are simulated, the code has been implemented so that it takes the last year as an approximation.

#### *3.1.1.3.. Planetary Ephemerides*

The ephemeris gives the trajectory of astronomical objects in space, as the position and the velocity of it during a certain time. They are computed from mathematical models and available by NASA's JPL HORIZONS system [48]. This ephemeris are useful for gravity perturbations that affect the satellite's motion. Sun and planetary ephemeris from the DE436 model are used which go from 1945 to 2150.

#### *3.1.1.4.. Earth Gravity field coefficients*

Gravity consideration of the Earth of $g = 9.81 m/s^2$ is an approximate value. As Earth is not a perfect sphere of uniform mass density its gravity field varies depending on the location. NASA's GRACE mission [41]measurements are obtained (model GGM03)and used to compute the Earth's gravity field variations, which affect the satellite's motion.

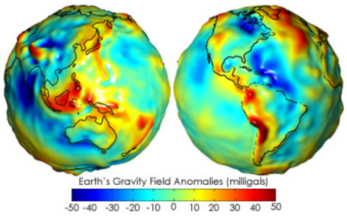



Figure 3.2: Representation of Earth's gravity field by GRACE mission

### 3.1.2.. Orbital Perturbations

In the first version of the simulator, perturbations are not taken into account, as the orbital propagator is based on linear variations of the mean motion. The most important one is the Earth's gravity, however other accelerations must be taken into account if a more approximated propagator is wanted.[3]

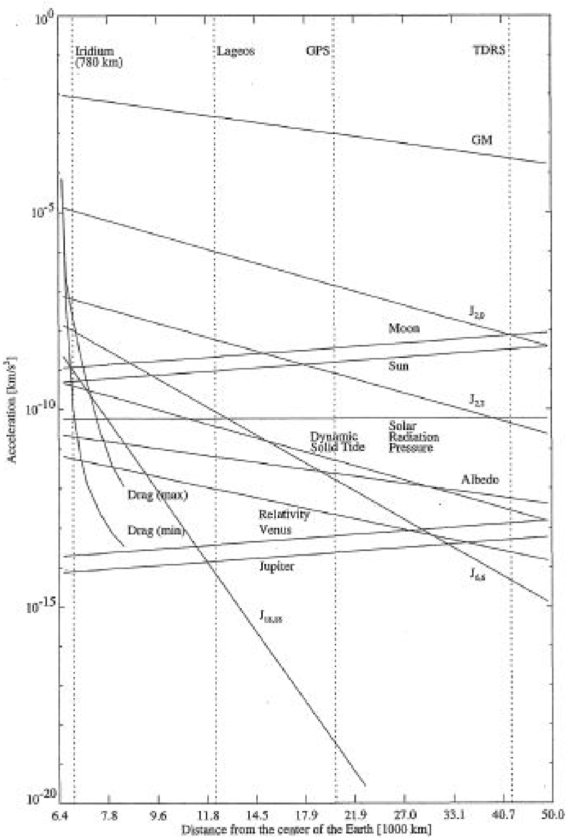



Figure 3.3: Representation of the perturbations importance to the orbiting satellite

#### *3.1.2.1.. Drag Acceleration*

Atmospheric drag at orbital altitude is caused by frequent collisions of gas molecules with the satellite. It is the major cause of orbital decay for satellites in low Earth orbit which are frequently used for Earth observation. As smaller is the satellite's orbital altitude, bigger is the atmosphere density so the drag perturbation increases.

An empirical thermospheric density model, Jacchia-Bowman 2008 [46]is implemented with an open source function to compute the atmosphere density depending on the satellite's position.

The acceleration on the satellite due to atmospheric drag is[7]:

$$a_D = \frac{F_D}{m_{sat}} = -\frac{C_D \times A_s at \times \rho \times V^2}{2 \times m_{sat}} \qquad [\mathrm{m/s^2}] \quad (3.1)$$

where ρ is the density obtained from the solar indices and geomagnetic storm data.

#### 3.1.2.2.. Gravity perturbations

In the first version of the simulator, only Earth's gravitation was considered. However, in reality Moon and Sun gravity effects also affect the motion of the satellite. With the ephemeris positions of the Sun, Moon and the Solar System planets, the gravitational perturbations are computed[7]:

$$a_G = -GM \frac{R_{sat} - R_M}{|R_{sat} - R_M|^3} - \frac{R_{sat}}{|R_{sat}|^3} \quad (3.2)$$

#### 3.1.2.3.. Solar Radiation Pressure

Solar radiation pressure is the pressure exerted upon any surface due to the exchange of momentum between the object and the electromagnetic field of the Sun's radiation. It has more effect on smaller bodies such as satellites as they have a larger ratio of surface area to mass. [7]

$$a_{SP} = -P_S \times C_R \times \frac{A_{sat}}{m_{sat}} \times AU^2 \times \frac{R_{sat} - R_{Sun}}{|R_{sat} - R_{Sun}|^3} \quad (3.3)$$

being $C_R$ the radiation pressure coefficient:

$$C_R = 1 + \varepsilon \quad (3.4)$$

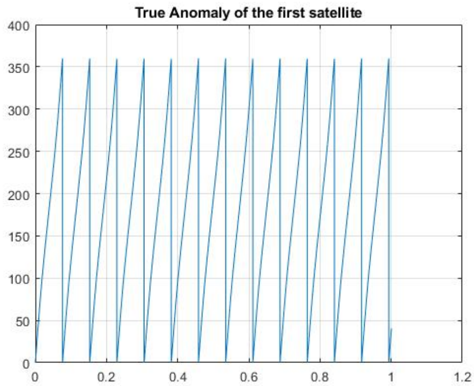
and $P_S = 4.56 \times 10^{-6} N/m^2$ the solar pressure.

### 3.1.3.. Results of the implementation

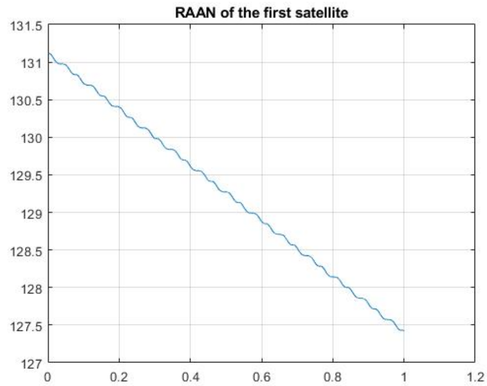
In order to check the reliability of the implementation, the same mission example as in the first version of the simulator explanation has been simulated.

The code has been computed so that the satellite initial position can be introduced either in TLE format or using Keplerian orbital parameters.

For an orbit at 550 km high and 50º inclination, propagated during 1 day, the following results are obtained:




(a) True anomaly evolution

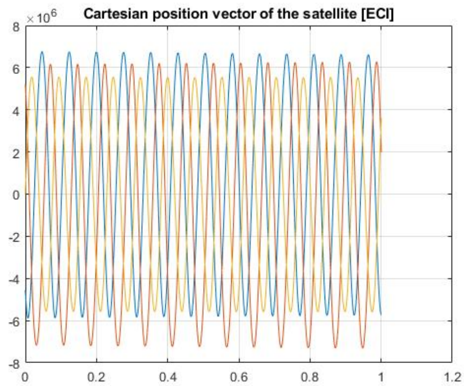



(b) RAAN evolution

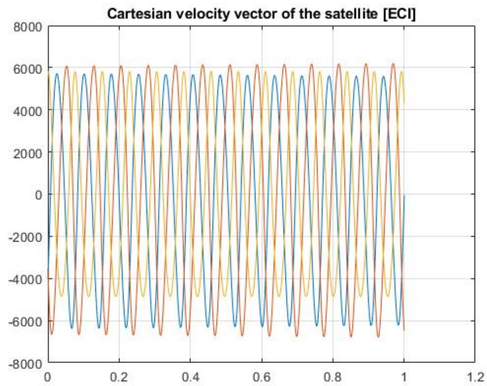
Figure 3.4: Keplerian parameters propagation

The main difference from the Keplerian elements evolution of the first version, is that the true anomaly of the satellite does not variate linearly due to the perturbations added, so this will make the satellite propagate a little slowly.

- ECI REFERENCE SYSTEM



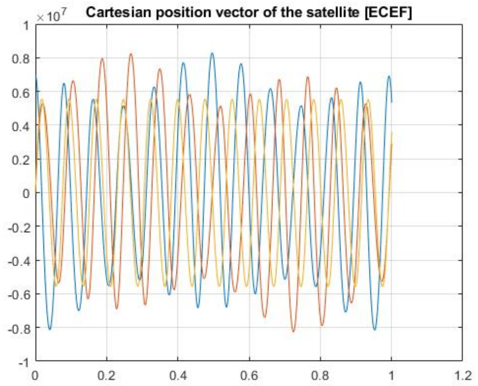

(a) ECI position



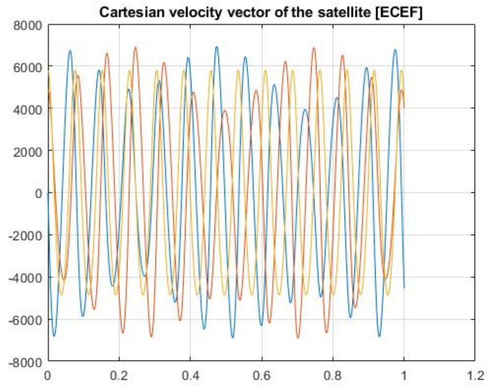

(b) ECI velocity

Figure 3.5: ECI computation

- ECEF REFERENCE SYSTEM



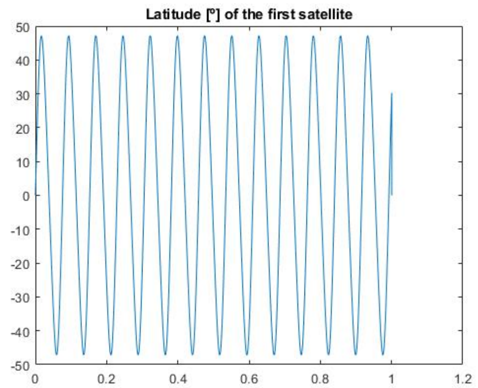

(a) ECEF position



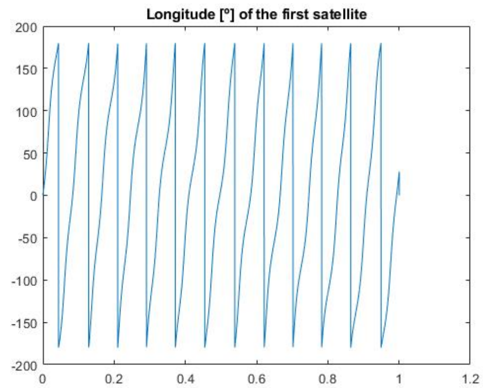

(b) ECEF velocity

Figure 3.6: ECEF computation

- GEOGRAPHICAL REFERENCE SYSTEM




(a) Latitude evolution




(b) Longitude evolution

Figure 3.7: Geographical computation

ECI and ECEF coordinate systems position and velocity do not show any significant changes, the only thing is that their values are more matched in the three axis.

From the geographical coordinates, the main difference is in the latitude values. It can be observed that the satellite does not reach the 50º of inclination. This is due to the perturbations added, so now the inclination values must be a little higher than the latitude to pass by that zones.

In the ground-track this difference can be seen, as the satellite does not reach the zones located at 50º.

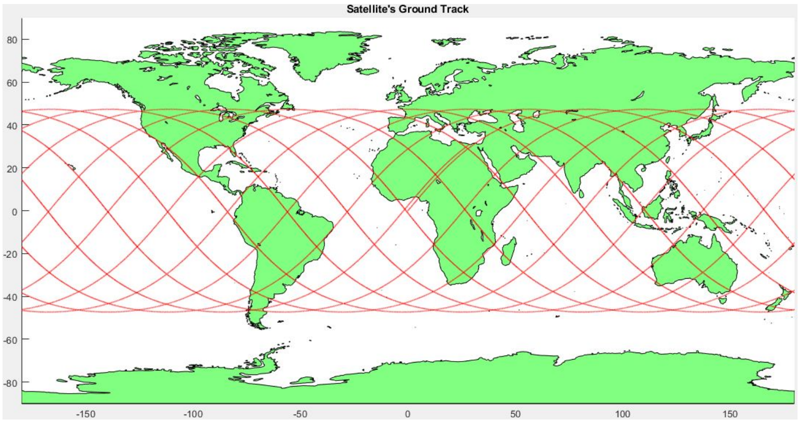
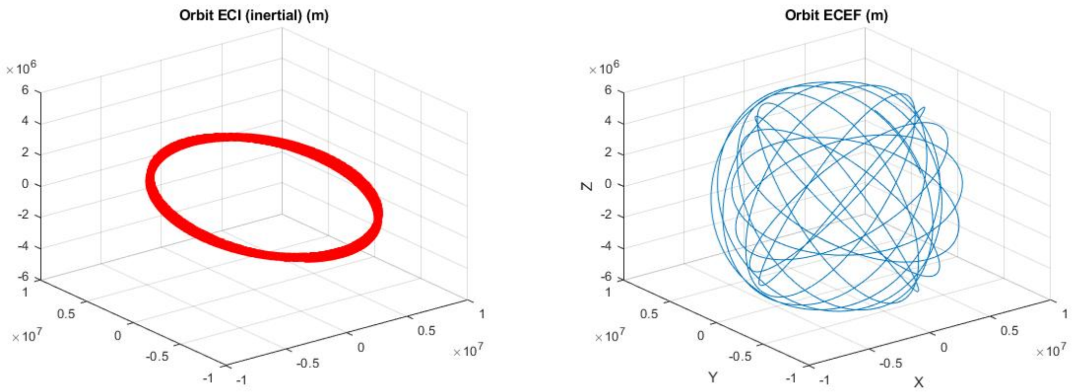



Figure 3.8: Ground-track for a 550 km orbit with $50^{o}$ of inclination during 1 day




(a) Orbit in ECI

(b) Orbit in ECEF

Figure 3.9: Propagation of the orbit respect the Earth's center

## 3.2.. Solar Radiation Constant Variation

The solar radiation constant is a flux density parameter that measures the mean solar electromagnetic radiation per unit area. It is measured on a surface perpendicular to the rays, one astronomical unit (AU) from the Sun, which is the distance from Sun to Earth. As it is the main power source to feed the satellite, it is a key parameter to take into account.

In the first version this is considered as a constant with a value of $S = 1400W/m^2$. However, that is not true as the Earth's orbit around the Sun is elliptic (e=0.01671022) and so, the distance between the Sun and Earth varies through the year.

To obtain a better approximation to reality, the orbit of the Earth and the sun is analyzed:

During the apogee, the minimum solar radiation will arrive to the satellite as Earth's is in the farthest point from Sun. This happens every year around July 5th. The same way on the perigee which occurs around January 4th, the solar radiation arriving to the satellite will be maximum.

A code has been designed so that depending on the day of the simulation with respect this apogee and perigee dates, the initial true anomaly $T_a$ of the Earth around the Sun is determined and so its initial position in the orbit.

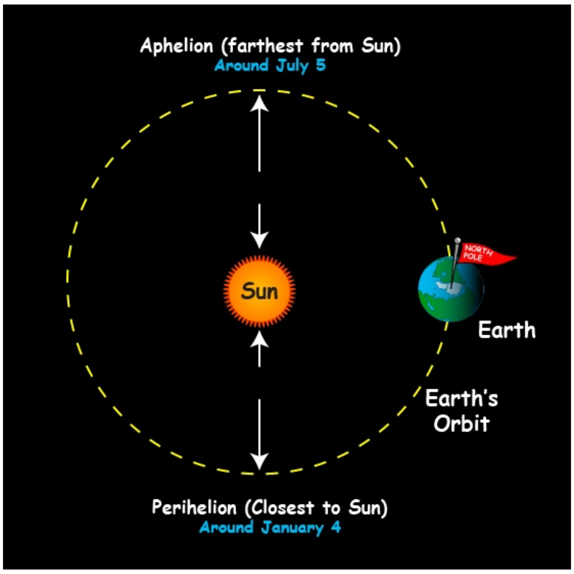



Figure 3.10: Earth's orbit around the sun with apogee and perigee (NASA source

Once this value is obtained, the eccentric anomaly is computed:

$$E_o = atan2\left(\frac{\sin(T_{a_o}) \times (1-e)^{0.5}}{\cos(T_{a_o}) + e}\right) \qquad \text{[rad]} \quad (3.5)$$

and then, the initial mean anomaly can be calculated:

$$M_a = E + e \times \sin(e) \qquad \text{[rad]} \quad (3.6)$$

Knowing the Earth's position on the Sun orbit allows us to compute the distance between both:

$$D_{ES} = a \times \frac{(1-e^2)}{1 + e\cos(T_a)} \qquad \text{[m]} \quad (3.7)$$

and so the solar constant value is:

$$S = \sigma \times T_{Sun}^4 \times (\frac{4\pi \times R_{Sun}}{4\pi \times D_{ES}})^2 \qquad \text{[W/m}^2\text{]} \quad (3.8)$$

In order to propagate the Earth, the angular velocity in each iteration is computed and added to the true anomaly. Firstly the linear velocity is calculated:

$$V_{ES} = \sqrt{2GM \times (\frac{1}{D_{SE} - \frac{1}{2a}})} \qquad \text{[m/s]} \quad (3.9)$$

Then the angular velocity:

$$w_{ES} = \frac{V_{ES}}{D_{SE}} \qquad \text{[rad/s]} \quad (3.10)$$

and finally it the anomaly is propagated in each iteration:

$$M_{a_i} = M_{a_i - 1} + w_{ES} \times t_{step} \qquad \text{[rad]} \quad (3.11)$$

To validate the code implementation, a 1-year simulation from the perihelion date is computed and the following results are obtained:

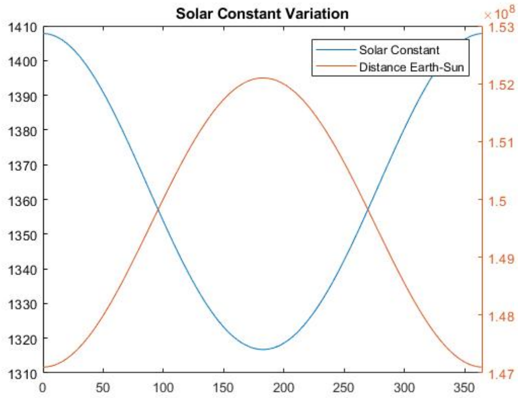



Figure 3.11: Solar constant variation in Watts per squared meter compared to the Earth-Sun distance in meters

It can be observed, in the first days of the simulation, as it started in the perigee where the distance is minimum, the solar constant is in its highest value of $S_{max} = 1408W/m^2$, while during the beginning of July when the distance is the highest, it achieves its lower values of $S_{min} = 1317W/m^2$.

This implies a variation of a 7% in the power that the satellite receives, which might be a key variation during any moment of a mission.

## 3.3.. Sun Safe Mode

Sun safe mode is a commonly used configuration for satellites which main objective is to ensure that the system remains safe.[43] This is achieved by:

- Maximum energy absorption
- Avoid exposing sensitive instruments to the solar radiation
- Thermal effect balance

The first version of the code simulated always the same orientation. Satellite is constantly pointing to the center of the Earth with face 1 (Z axis), where the camera is located so that Earth observation could be achieved, while face 2 (x-axis) pointed towards the satellite's orbit velocity.

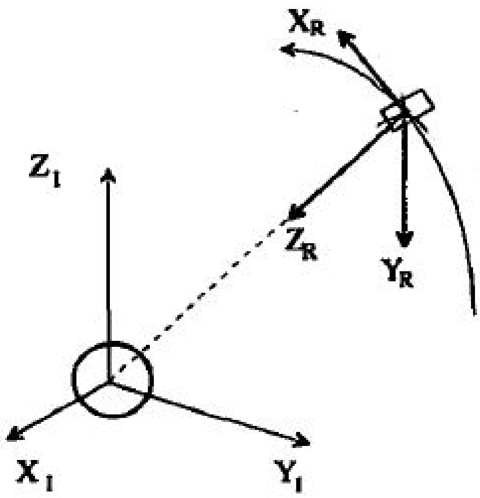



Figure 3.12: Representation of the standard simulated orientation

To satisfy the first constraint, the orientation of the satellite must be changed so that the maximum number of solar panels absorb solar radiation. The satellite-6U simulated has the following size characteristics:

The bigger area, the more satellite panels can be placed in that face and so, more power is acquired. That is why face 2 or face 5 whose normal vector is the x-axis is forced to point directly to the Sun. If face 2 points the Sun

$$N_{A2} = [X_{Sun} - X_{sat}, Y_{Sun} - Y_{sat}, Z_{Sun} - Z_{sat}] \tag{3.12}$$

Then face 5 points always to the opposite side where the Sun is. Its normal vector is:

$$N_{A5} = -[X_{Sun} - X_{sat}, Y_{Sun} - Y_{sat}, Z_{Sun} - Z_{sat}] \tag{3.13}$$

To compute the normal vectors of the adjacent faces 3 and 4, an Euler rotation matrix around the x axis of $\phi = 90^o$ is applied:

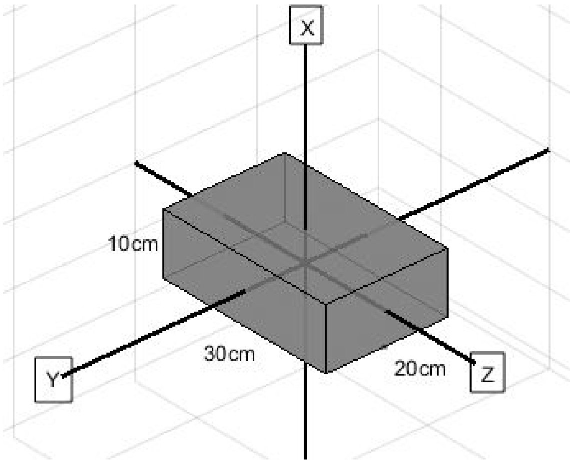



Figure 3.13: Representation of the satellite body axis

Counter clock wise rotation for face 3:

$$\begin{pmatrix} N_{A3X} \\ N_{A3Y} \\ N_{A3Z} \end{pmatrix} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & cos(\phi) & -sin(\phi) \\ 0 & sin(\phi) & cos(\phi) \end{bmatrix} \begin{pmatrix} N_{A2X} \\ N_{A2Y} \\ N_{A2Z} \end{pmatrix}$$

Clock wise rotation for face 4:

$$\begin{pmatrix} N_{A4X} \\ N_{A4Y} \\ N_{A4Z} \end{pmatrix} = - \begin{bmatrix} 1 & 0 & 0 \\ 0 & cos(\phi) & -sin(\phi) \\ 0 & sin(\phi) & cos(\phi) \end{bmatrix} \begin{pmatrix} N_{A2X} \\ N_{A2Y} \\ N_{A2Z} \end{pmatrix}$$

Finally to obtain the faces 1 and 6 face's normal vectors, a cross product with faces 2 and 4 is calculated such that:

$$\begin{pmatrix} N_{A1X} \\ N_{A1Y} \\ N_{A1Z} \end{pmatrix} = N_{A2} \times N_{A4} = \begin{bmatrix} \hat{\imath} & \hat{j} & \hat{k} \\ N_{A2X} & N_{A2Y} & N_{A2Z} \\ N_{A4X} & N_{A4Y} & N_{A4Z} \end{bmatrix}$$

$$\begin{pmatrix} N_{A6X} \\ N_{A6Y} \\ N_{A6Z} \end{pmatrix} = N_{A4} \times N_{A2} = \begin{bmatrix} \hat{\imath} & \hat{j} & \hat{k} \\ N_{A4X} & N_{A4Y} & N_{A4Z} \\ N_{A2X} & N_{A2Y} & N_{A2Z} \end{bmatrix}$$

Once all normal vectors are obtained, the same procedure as in the initial simulator version is done in order to project the areas respect to the Sun's vector and obtain the areas projection.

However, if the satellite's face 2 stays always pointing to the Sun receiving the maximum possible radiation, its parallel face which is face 5 is always in the dark. This creates a thermal unbalance. While face 2 receives permanent radiation and so the temperature of the equipment near it rises hugely, face 5 will only release heat to space without receiving, and in consequence the equipment placed near this face will evolve to low temperatures.

It is known that the equipment life and performance depends on its temperature, that is why a thermal balance system needs to be implemented.

This thermal balance system consists in a continuous rotation around the faces 1 and 6 normal axis. In this way, the normal vectors of faces 2, 3, 4 and 5 are continuously rotating with a concrete spin angular speed, so that its projection area varies from its maximum to minimum values.

Computing this rotation requires the implementation of the Euler-Rodrigues theorem [22].

The Rodrigues' formula is an algorithm that, given an axis $k$ and an angle $\theta$ of rotation $\theta$ it rotates a desired vector $v$ in space, obtaining a new rotated vector $v_{rot}$:

$$V_{rot} = v\cos\theta + (k \times v)\sin\theta + k(kv)(1-\cos\theta) \tag{3.14}$$

To implement this formula to our problem:

$$k = [N_{A1X}, N_{A1Y}, N_{A1Z}]$$

and the rotated vectors $v$ are those of the faces 2, 3, 4 and 5:

$$v = \begin{pmatrix} N_{A2X} & N_{A3X} & N_{A4X} & N_{A5X} \\ N_{A2Y} & N_{A3Y} & N_{A4Y} & N_{A5Y} \\ N_{A2Z} & N_{A3Z} & N_{A4Z} & N_{A5Z} \end{pmatrix} \tag{3.15}$$

As a continuous rotation with an angular velocity $w_{spin}$ is required for our problem, the angle of rotation must be incremented in every iteration such that:

$$\theta = w_{spin} \times t_{step} \qquad [\text{rad}] \tag{3.16}$$

In this way, the constraint of maintaining a thermal balance is achieved. The heat transfer between the 4 faces of the satellite is matched, therefore any part of the satellite is not damaged due to extreme temperature conditions.

To achieve the objectives of this mode, it has been implemented in the scheduler under some conditions. If the battery capacity levels of the satellite are compromised, this safe mode will be activated until the batteries are charged. While this mode is working, and the satellite will stop Earth observation as the orientation of the camera is not the desired one.

#### *3.3.0.1.. Results and validation*

To check the viability of this mode, the projected areas and power received of the satellite must be validated. A simulation during one day has been done with the same conditions as always, comparing the standard mode and sun safe mode. The following plots show the results obtained:

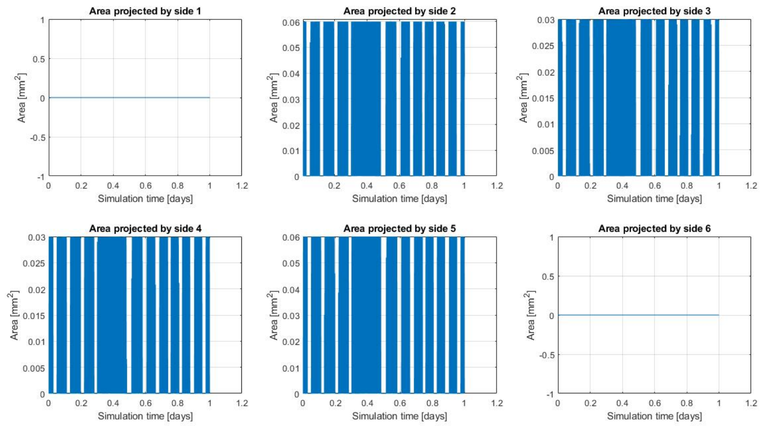



Figure 3.14: Projection of the satellite areas to Solar radiation during 1 day

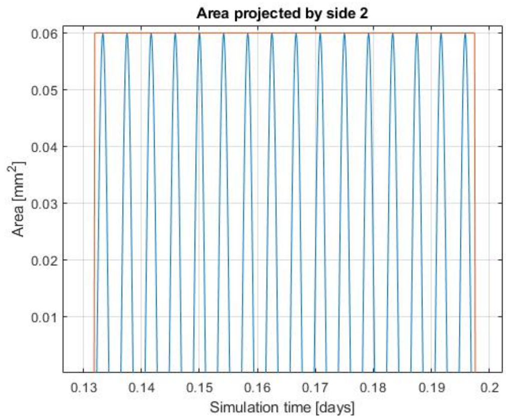



Figure 3.15: Amplified vision of face 2 area's projection.

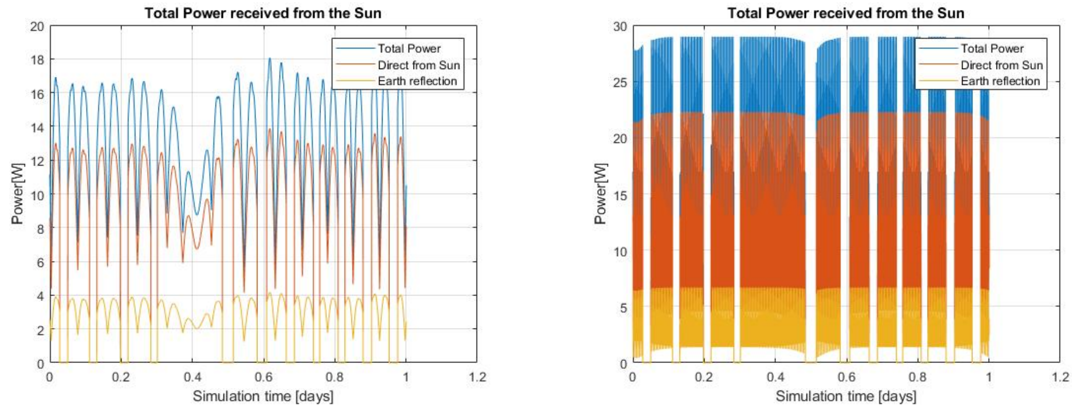
The first thing to notice is that faces 1 and 6 do not receive solar radiation as they are always perpendicular to the Sun vector, so the second constraint of avoiding exposing sensitive instruments to the solar radiation is achieved, as the camera is located in face 1.

As expected, the four other faces will receive the solar radiation. If a zoom is made to the plots, in the figure 3.15, it can bee seen how when the normal vector of the satellite is parallel with the direction of the sunlight, the maximum projected area is achieved so that

$$A_{2projected} = A_{2face} = 0.06m^2 \tag{3.17}$$

and along the rotation this value fluctuates through time from its minimum $(0m^2)$ to its maximum, depending on the angle between the satellite's face and the sunlight direction (2.25).

About power received, in the following plots the notorious difference can be observed. While the standard mode absorbs a mean of 17 W, in sun safe mode up to 28 W are obtained.

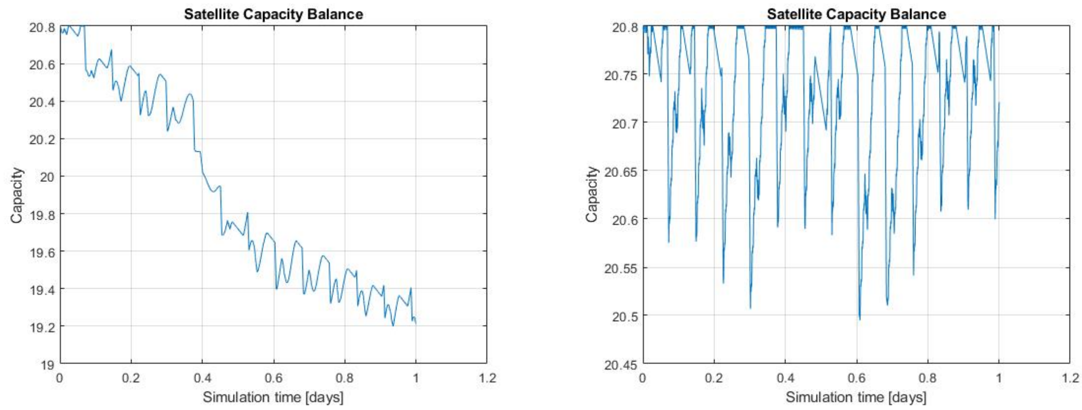



(a) Power received in the standard mode

(b) ECI velocity

Figure 3.16: Power acquired comparison between standard and Sun safe modes

This power is used to feed the batteries, so there is also an important difference in its capacity. While in the standard mode during 1 day the battery goes down to 19 Ah, in Sun safe this loss is recovered and batteries are almost always full charged.

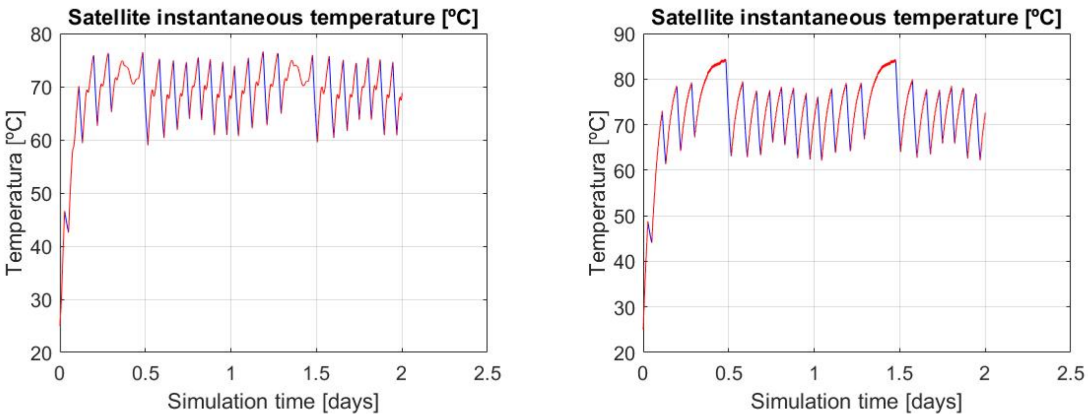



(a) Battery capacity (Ah) in standard mode

(b) Battery capacity (Ah) in sun safe mode

Figure 3.17: Comparison of battery charge between both modes during 1 day simulation

However, this mode can not be activated during large periods of time as the temperature rises due to the radiation absorbed, which may affect the performance of equipment. So there needs to be a balance in the use of the different modes. In the following figures, the difference of temperature between both is shown in a simulation:




(a) Satellite's temperature in standard mode (b) Satellite's temperature in sun safe

Figure 3.18: Comparison of the temperatures between both modes during 2 day simulation

## 3.4.. Mission Lifetime

The mission lifetime of a satellite is a parameter that measures the amount of time required for the satellite to deorbit under the influence of atmospheric drag alone.
This drag force reduces the velocity insignificantly in short time periods, but with the time this speed change becomes notorious and so the orbit altitude starts decreasing and drag increases due to the atmosphere proximity. The end of this altitude decay process is to experience a re-entry in atmosphere and burn.[47]

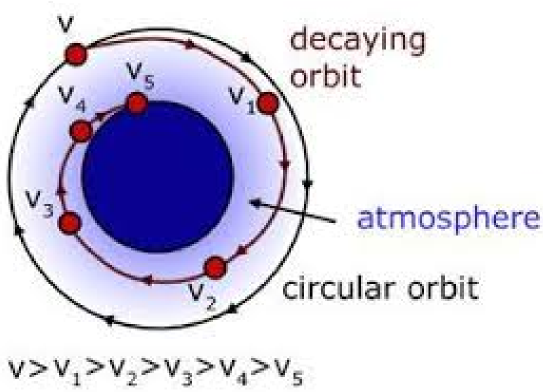



Figure 3.19: Representation of an orbit decay for circular orbit

This parameter gains importance when the satellite used for the mission is in lower altitudes, such as the LEO for Earth observation.

A function has been implemented so that, depending on the initial altitude of the orbit designed to fulfill the mission, a lifetime value is obtained.

Firstly, the orbit initial altitude is computed:

$$h = a - R_E \qquad [\mathrm{m}] \quad (3.18)$$

Depending on the initial altitude, a scale height H value and the mean density ρ is obtained from databases of our atmosphere [27] [45]. The data used has been acquired from: http://www.braeunig.us/space/atmos.htm.

Then the ballistic coefficient of the satellite is computed, which is a measure of a body's ability to overcome air resistance in flight:

$$B_c = \frac{m_{sat}}{C_D \times A_{max}} \qquad [\mathrm{kg/m^2}] \quad (3.19)$$

where the maximum cross area has been taken to consider the worst case. The drag coefficient used is $C_D = 2$ [39] as it is the common average value for satellites in low orbits.

Once the ballistic coefficient of the satellite is obtained, the changes in semi-major axis per revolution are:

$$Da_{rev} = -2\pi \times \frac{\rho \times a^2}{B_c} \qquad [\mathrm{m}] \quad (3.20)$$

and so, the number of orbits that the satellite performs before crashing into the atmosphere are:

$$N_{orbits} = -\frac{H}{Da_{rev}} \quad \text{[orbits]} \quad (3.21)$$

A lifetime time value of the satellite can be obtained by using the period T of our satellite:

$$Lifetime = \frac{N_{orbits}}{86400} \times 2pi\sqrt{\frac{a^3}{GM}} \quad \text{[years]} \quad (3.22)$$

For example, for a 550km altitude orbit, the lifetime obtained is about 16 years. However if this orbit decreases up to 500km, the mean lifetime obtained is about 6 years and 10 months, which is more than half less.

Therefore, this is a very important parameter to take into account when designing a satellite mission for LEO orbits as it will limit the years of use.

Of course, this lifetime expectancy can be increased by using propellers in the satellite. They are used to get impulse and recover the velocity loss from the atmosphere or space perturbations and so increase again the orbit altitude.

## 3.5.. GomSpace real equipment implementation

GomSpace [9] is a manufacturer and supplier of nanosatellites for customers in the academic, government and commercial markets. It provides satellite's equipment and its data. In order to make the simulator more realistic, the different subsystems used in the simulator have been implemented with the data of this real equipment.

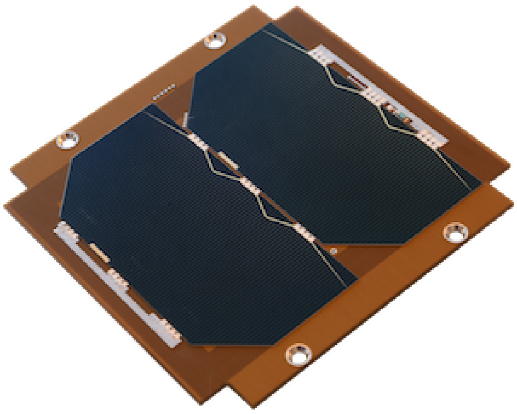
Depending on the user election of the size of the satellite ( 1U, 2U, 3U or 6U), different kind of equipment can be selected and so its main characteristics such as consume, intensity or performance is implemented to the simulator software.

The available equipment implemented to use on the satellite simulated are:

Lithium Ion 18650 cell

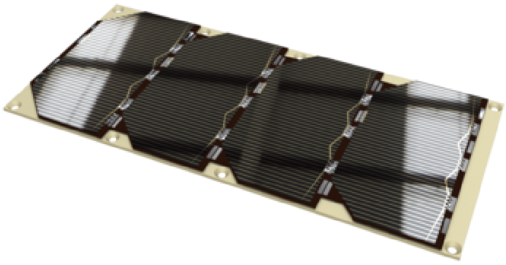
- Voltage = 3.7 V
- Capacity = 2600 mAh
- DOD 75 / 50 / 100 % =1700 / 1000 / 350 cycles

NanoPower P110 Solar Panel [15]



- Efficiency = 30 %
- Cell Area = 60.36 $cm^2$
- Panel Area = 80.5 $cm^2$
- Effective Area = 74.5 %
- Maximum Power = 2.3 W

NanoPower MSP Solar Panel[14]

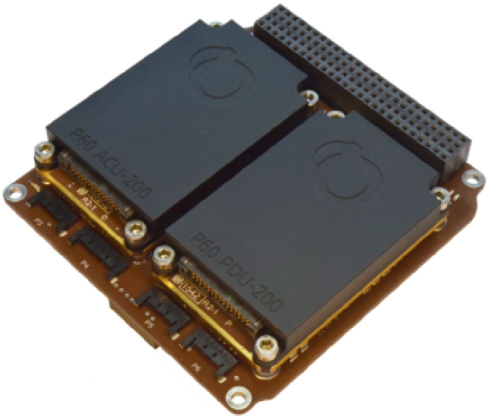


- Efficiency = 30 %
- Cell Area = 30.62 $cm^2$
- Panel Area = 40 $cm^2$
- Effective Area = 75 %
- Maximum Power = 1.15 W

Electric Power Supply system (EPS) P31U[11]

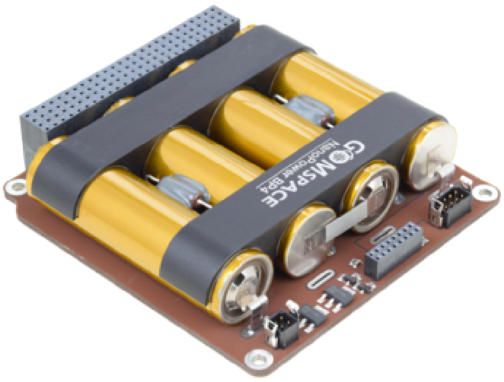

- Efficiency = 87.5 %
- Voltage = 5 V
- Current = 4 A
- Maximum Power = 30 W
- Integrated battery: 2 cells, 2.6 Ah, 8 V
- Heater: 22 Ω, 3 W

Electric Power Supply system (EPS) P60[10]



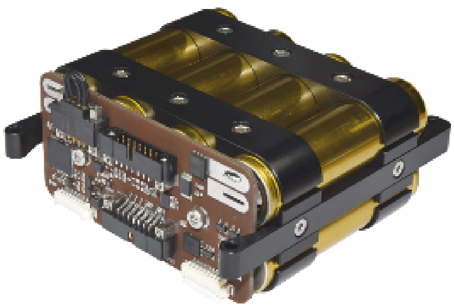

- Efficiency = 88 %
- Voltage = 20
- Current = 4 A

NanoPower BP4 Battery[13]




- 4 cells (2 series - 2 parallel)
- Voltage = 7.2 V
- Capacity = 5.2 Ah
- Heater: 22Ω, 6 W

NanoPower BPX Battery[12]

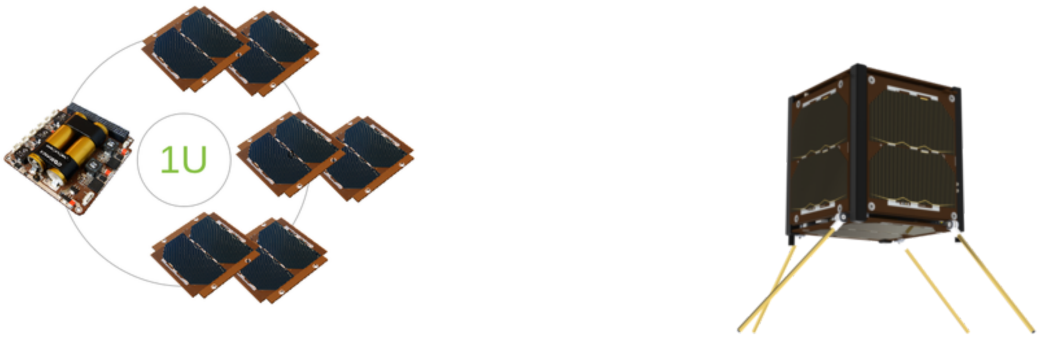


- 8 cells (2 series - 4 parallel)
- Voltage = 7.4 V
- Capacity = 10.4 Ah
- Heater: 10Ω, 6 W

With this new equipment, the following satellite configurations can be done:

1U Satellite [18] [21]

- EPS P31u
- 5 P110 Solar Panels
- P31u Battery




(a) Equipment - 1U

(b) Satellite - 1U

Figure 3.20: Representation of a satellite of size 1U

2U Satellite [17]

- EPS P31u
- 9 P110 Solar Panels
- BP4 Battery

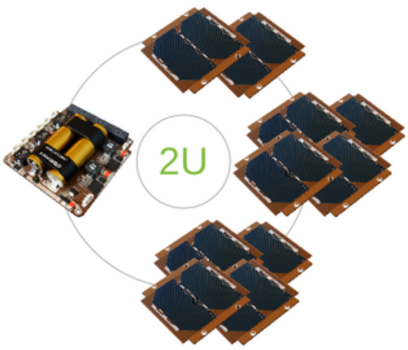
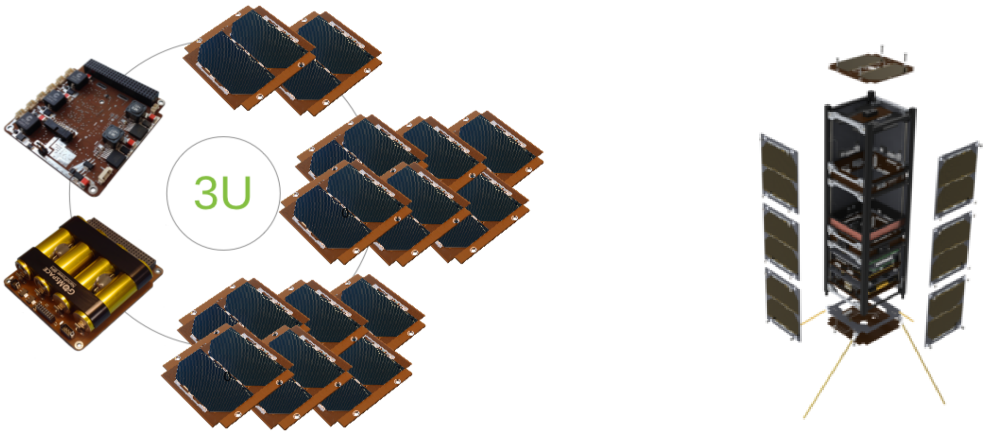



Figure 3.21: Equipment - 2U

3U Satellite [16] [20]

- EPS P31u
- 13 P110 Solar Panels
- BP4 Battery and P31u Battery




(a) Equipment - 3U (b) Satellite - 3U

Figure 3.22: Representation of a satellite of size 3U

6U Satellite [19]

- EPS P60
- 51 cells of MSP Solar Panels
- BPX Battery

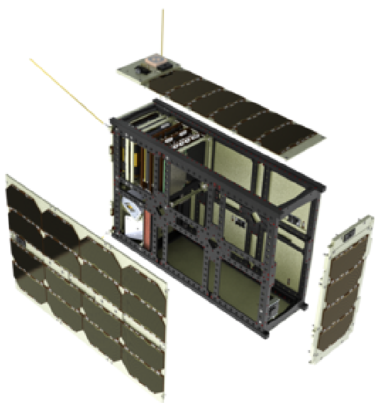


Figure 3.23: Satellite of size 6U

## 3.6.. Open Cosmos Ground Stations Implementation

The localization of ground stations for the mission design is a very important parameter to take into account as a good positioning of them will allow the satellite to take more photos or to download faster the collected data.

The first version of the simulator did not include any specific location of ground stations. Only its position was needed to be introduced by the user.

In this new version, 23 ground stations data have been updated to the software. This ground stations are real ones which are actively working nowadays. This will allow the user to make a more realistic mission simulation as the position of the different stations is shown, and therefore the user can select which ones are the best to use or also change its orbital design due to the absence of ground stations through the ground track first designed.

In the following table, a list of the new implemented ground stations is shown:

| Ground Station List | | | |
|---|---|---|---|
| NAME and CODE | LATITUDE | LONGITUDE | ALTITUDE |
| Esrange ESR | 67.88$^o$ | 21.07$^o$ | 341 m |
| Inuvik INU | 68.40$^o$ | -133.50$^o$ | 51 m |
| North Pole NP | 64.80$^o$ | -147.65$^o$ | 135 m |
| Clewiston CLE | 26.73$^o$ | -82.03$^o$ | 3 m |
| South Point SPO | 19.02$^o$ | -155.67$^o$ | 164 m |
| Yatharagga YAT | -29.08$^o$ | 115.58$^o$ | 280 m |
| Dongara DON | -29.05$^o$ | 115.35$^o$ | 280 m |
| Santiago STG | -33.13$^o$ | -70.67$^o$ | 698 m |
| Punta Arenas PAR | -52.93$^o$ | -70.85$^o$ | 88 m |
| Fucino FUC | 42.00$^o$ | 13.55$^o$ | 652 m |
| Hartebeesthoek HAR | -25.64$^o$ | 28.08$^o$ | 1288 m |
| Svalbard SVA | 78.23$^o$ | 15.41$^o$ | 248 m |
| TrollSat TRO | -72.02$^o$ | 2.53$^o$ | 1270 m |
| Tromso TMS | 69.39$^o$ | 18.56$^o$ | 4 m |
| Grimstad GRI | 58.34$^o$ | 8.59$^o$ | 28 m |
| Puertollano PLL | 38.69$^o$ | -4.11$^o$ | 703 m |
| Singapore SGP | 1.35$^o$ | 103.82$^o$ | 55 m |
| Mauritius MAU | -20.35$^o$ | 57.55$^o$ | 579 m |
| Panama PAN | 8.54$^o$ | -80.78$^o$ | 1057 m |
| Fairbanks FBA | 64.80$^o$ | -147.70$^o$ | 135 m |
| Dubai DUB | 25.20$^o$ | 55.27$^o$ | 0 m |
| Hartebeesthoek HAR2 | -25.64$^o$ | 28.08$^o$ | 1288 m |
| Inuvik INU | 68.40$^o$ | -133.50$^o$ | 51 m |

Table 3.1: Positions of the ground stations

And they are also shown on ground-track when activated:

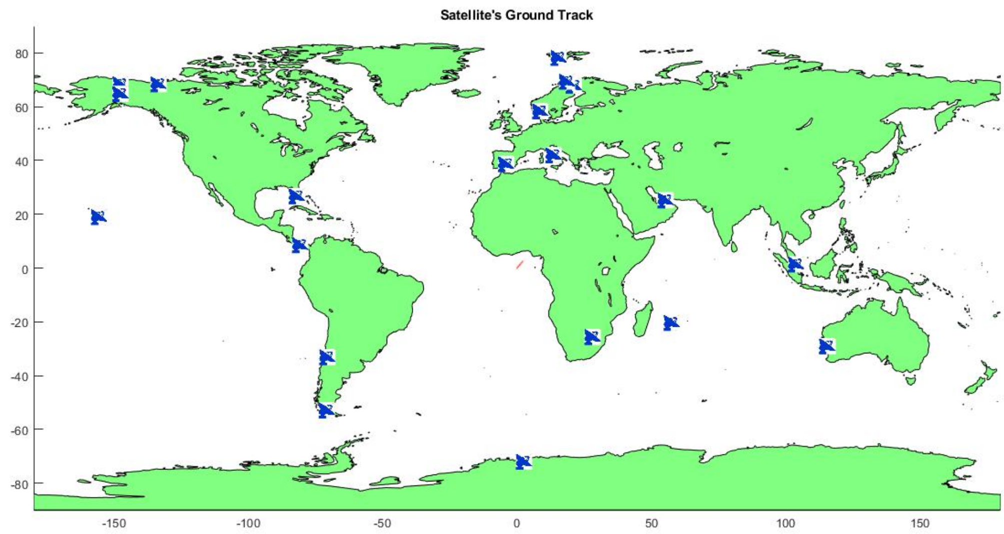



Figure 3.24: Representation of the available ground stations location on Earth's map

# CHAPTER 4. MISSION ANALYSIS AND DESIGN FOR EARTH OBSERVATION

Along this chapter, an example of future mission design for Earth observation is developed with a defined objective. The satellite simulator software developed and studied along the project is used to define the satellite system characteristics. Depending on the mission objective and constraints, different orbit design, equipment or subsystems characteristics will be chosen in order to achieve the best possible configuration for a future possible mission.

## 4.1.. Mission Objective and Requirements

The main objective of an Earth observation mission is to observe a geographical zone for collecting data.
In this example of mission, the zone of Catalonia will be studied. The mission objective is to collect data by taking images with the payload camera on board. The target area studied is Catalonia, data will be collected by the satellite and then interpreted on ground for multiple needs:

- Natural resources management
- Human impact in agriculture, forests, natural parks and geology.
- Meteorology phenomena observation
- Fire motorization
- Land observation, including: vegetation, soil and water cover and coastal areas

Thanks to the images achieved from the satellite point of view, this list of multiple purposes that affect our studied territory can be solved in a more sustainable way and also some future problems could be predicted.

In order to fulfill this objectives, the following constraints are set:

- Expected average time gap 7 days or less.
- Expected revisit time 2 to 5 days.
- Expected lifetime of at least 7 years.

### 4.1.1.. Target Area Studied - Catalonia

Catalonia is located in the west part of Europe, concretely in the Eastern part of Spain, under France. In geographical coordinates, the maximum and minimum latitudes are:

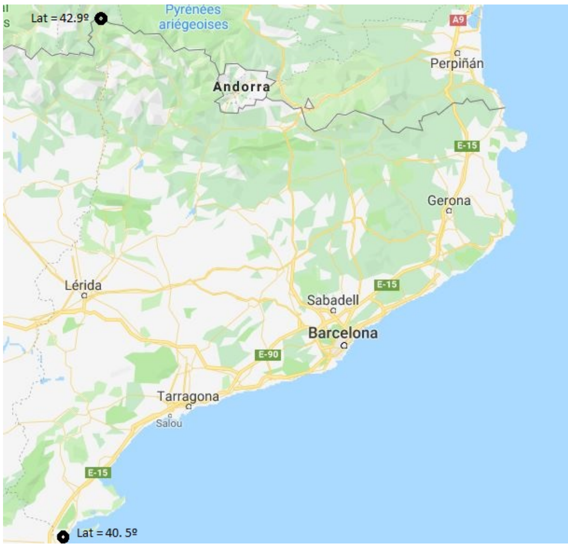



Figure 4.1: Catalonia's map with maximum and minimum latitudes

To introduce the data to the simulator app, maximum and minimum latitudes and longitudes are defined:

- Latitudes: from 40.5º to 42.9º
- Longitudes: from 0.3º to 3º

## 4.2.. Orbit Design

For Earth observation purposes, a LEO orbit is chosen. Low altitude orbits have numerous advantages: good resolution images, reduced energy budget required for communications and reduced cost of launch. However, the field of view when communicating is smaller and also space debris might be a problem.

As the maximum latitude of the orbit is 42.9º, the inclination angle should be at least equal to this number.

If an orbit is propagated with an inclination value equal to the latitude, the maximum latitude reached during the orbit won't arrive to the desired one as seen in the following figure. Maximum latitude value will be of 40.57º and the satellite won't be able to make photos:

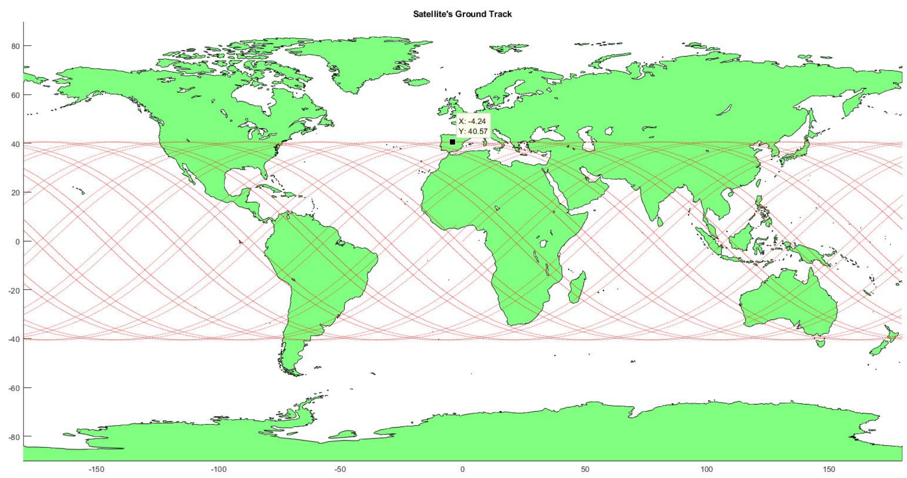



Figure 4.2: Orbit propagation of i = 40.57$^{o}$

Therefore, the initial inclination should be at least i = 45.4$^{o}$, so that latitudes of 42.9$^{o}$ are achieved.

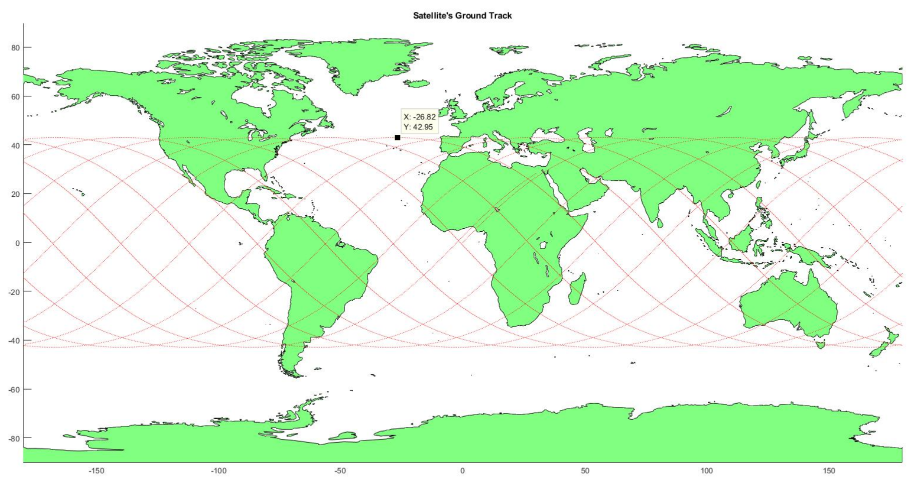



Figure 4.3: Orbit propagation of i = 45.4$^{o}$

Thanks to the Global Coverage function, the amount of times that the satellite passes by the different parts of the orbit can be obtained. For the last orbit propagated of i =45.4$^{o}$, the following result is obtained for a simulation of 10 days, and then analyzed:

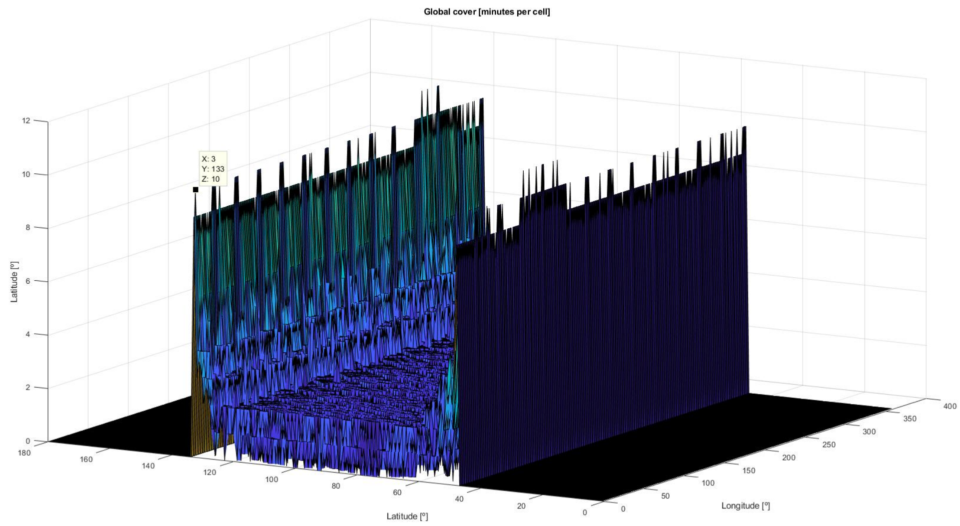



Figure 4.4: Global coverage for a satellite in an orbit of i = 45.4$^{o}$

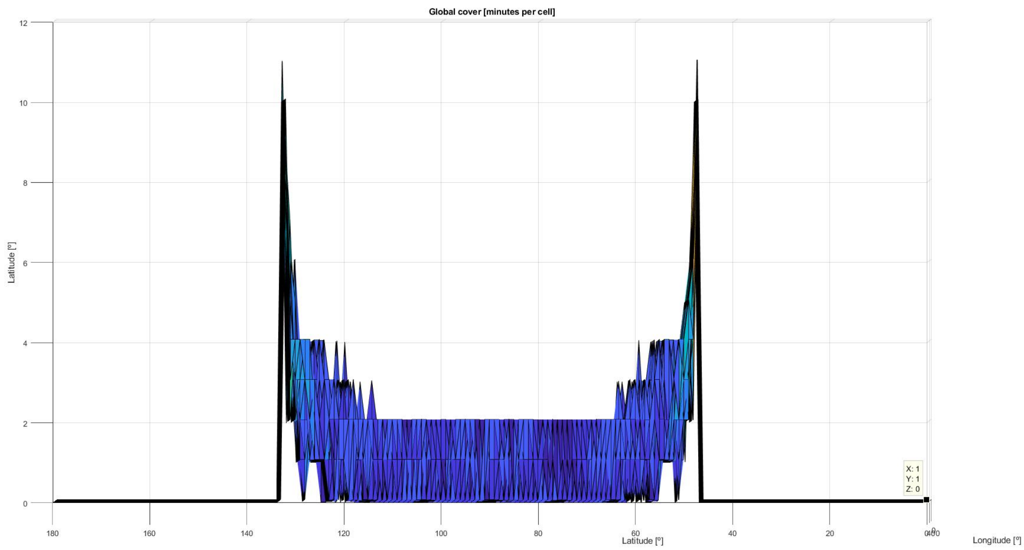



Figure 4.5: Global coverage for a satellite in an orbit of i = 45.4$^{o}$ lateral view

It can be observed, that while intermediate latitudes are visited from 4 to 2 times during the 10 days simulated, maximum and minimum latitudes are visited around 10 times. This means that the optimal inclination for the orbit should be its maximum, that is of i=45.4$^{o}$, in order to pass through Catalonia as frequently as possible.

Once the inclination is fixed, the next parameter to take into account is the altitude of the orbit. The main constraints that the altitude provides depending on its value its the lifetime and the coverage. Lower altitudes will provide more coverage to download data and better image resolution, but the satellite will induce more drag and so its lifetime may be compromised.

The minimum altitude can be computed by the restriction of at least 7 years of lifetime.

Several LEO lifetimes have been computed to compare:

| | **Altitude [km]** | **Lifetime** |
|---|---|---|
| orbit 1 | 475 | 4 years 1 month |
| orbit 2 | 500 | 6 years 10 months |
| orbit 3 | 525 | 9 years 9 months |
| orbit 4 | 550 | 23 years |

Table 4.1: Comparison of several LEO's lifetime

Therefore, the altitude of the orbit should be between 500km and 525km so that the lifetime is not compromised while keeping good resolution and coverage ranges.

## 4.3.. Ground Stations Coverage

Positioning accurately the ground stations in relation with our orbit is a key parameter of the mission. If an access data of x days is required, several ground stations must be activated. Observing the following ground track with the position of the ground stations on the map, some ground stations are clearly out of range due to its latitude position (in red) and so, the others are possible ground stations to use (green) . In the following tables, the list of ground stations is shown:

| Not viable GS List | | | |
|---|---|---|---|
| NAME and CODE | LATITUDE | LONGITUDE | ALTITUDE |
| Esrange ESR | 67.88$^o$ | 21.07$^o$ | 341 m |
| Inuvik INU | 68.40$^o$ | -133.50$^o$ | 51 m |
| North Pole NP | 64.80$^o$ | -147.65$^o$ | 135 m |
| Punta Arenas PAR | -52.93$^o$ | -70.85$^o$ | 88 m |
| Svalbard SVA | 78.23$^o$ | 15.41$^o$ | 248 m |
| TrollSat TRO | -72.02$^o$ | 2.53$^o$ | 1270 m |
| Tromso TMS | 69.39$^o$ | 18.56$^o$ | 4 m |
| Grimstad GRI | 58.34$^o$ | 8.59$^o$ | 28 m |
| Fairbanks FBA | 64.80$^o$ | -147.70$^o$ | 135 m |
| Inuvik INU | 68.40$^o$ | -133.50$^o$ | 51 m |

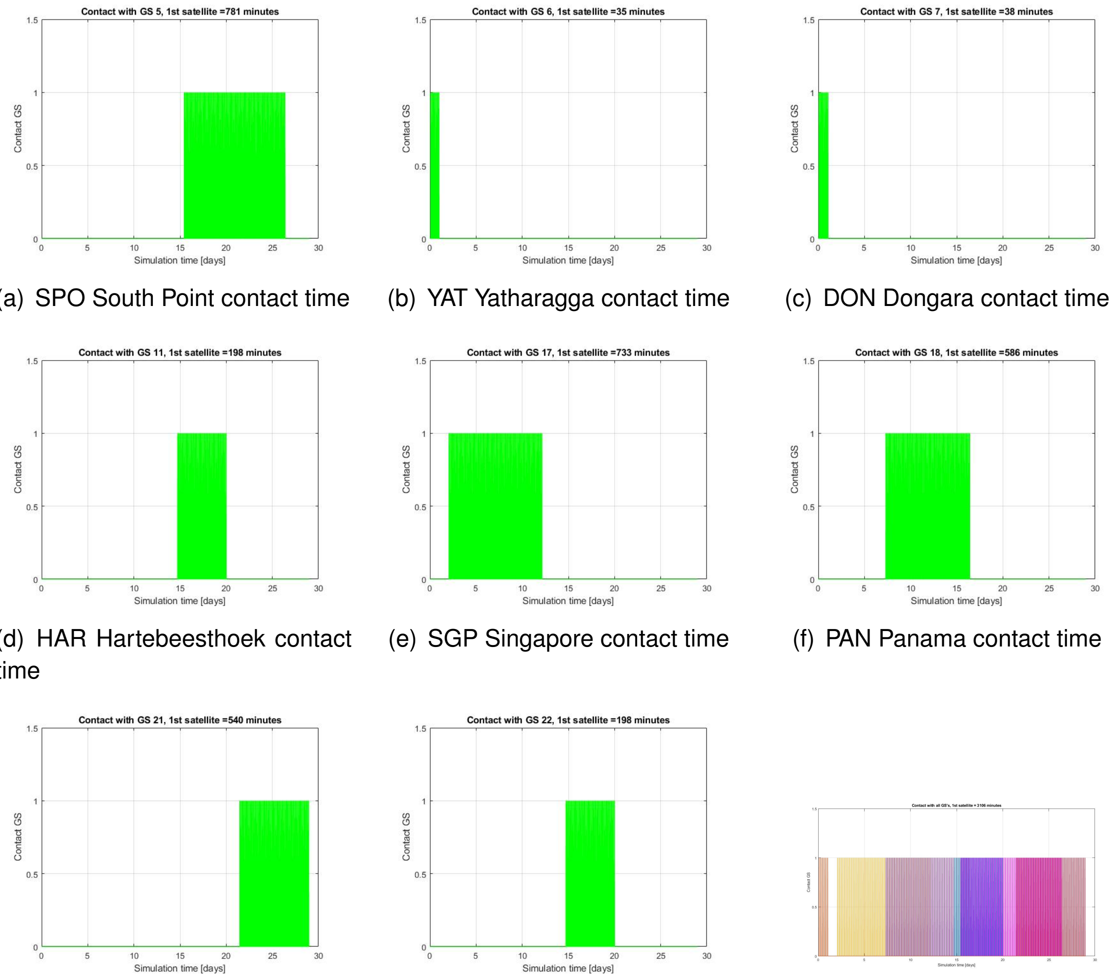
Table 4.2: Lists of ground stations that can not be used to download data

| Viable GS List | | | |
|---|---|---|---|
| NAME and CODE | LATITUDE | LONGITUDE | ALTITUDE |
| Clewiston CLE | 26.73$^o$ | -82.03$^o$ | 3 m |
| South Point SPO | 19.02$^o$ | -155.67$^o$ | 164 m |
| Yatharagga YAT | -29.08$^o$ | 115.58$^o$ | 280 m |
| Dongara DON | -29.05$^o$ | 115.35$^o$ | 280 m |
| Santiago STG | -33.13$^o$ | -70.67$^o$ | 698 m |
| Fucino FUC | 42.00$^o$ | 13.55$^o$ | 652 m |
| Hartebeesthoek HAR | -25.64$^o$ | 28.08$^o$ | 1288 m |
| Puertollano PLL | 38.69$^o$ | -4.11$^o$ | 703 m |
| Singapore SGP | 1.35$^o$ | 103.82$^o$ | 55 m |
| Mauritius MAU | -20.35$^o$ | 57.55$^o$ | 579 m |
| Panama PAN | 8.54$^o$ | -80.78$^o$ | 1057 m |
| Dubai DUB | 25.20$^o$ | 55.27$^o$ | 0 m |
| Hartebeesthoek HAR2 | -25.64$^o$ | 28.08$^o$ | 1288 m |

Table 4.3: Lists of ground stations that can be used to download data

By observing the Global Cover map computed before, it is known that the satellite will stay more time in latitudes near the orbit inclination, which means more time to download images. So, preferably, ground stations near latitudes of 40$^o$ may be chosen.

By running a simulation, the following results are obtained:




(a) SPO South Point contact time

(b) YAT Yatharagga contact time

(c) DON Dongara contact time

(d) HAR Hartebeesthoek contact time

(e) SGP Singapore contact time

(f) PAN Panama contact time

(g) DUB Dubai contact time

(h) HAR2 Hartebeesthoek contact time

(i) All GS contact time

Figure 4.6: Ground stations contact

As the mean access time required is of 7 days, by using 2 ground stations quite separated between them, to download data this value will be achieved.

Taking into account the simulation results of GS contacts , the final selection is:

| Selected GS List | | | |
|---|---|---|---|
| NAME and CODE | LATITUDE | LONGITUDE | ALTITUDE |
| South Point SPO | 19.02$^o$ | -155.67$^o$ | 164 m |
| Singapore SGP | 1.35$^o$ | 103.82$^o$ | 55 m |

Table 4.4: Lists of ground stations that can be used to download data

## 4.4.. Revisit Time

The next constraint to take into account is the revisit time. If a simulation of days is run, the following contacts along time are obtained.

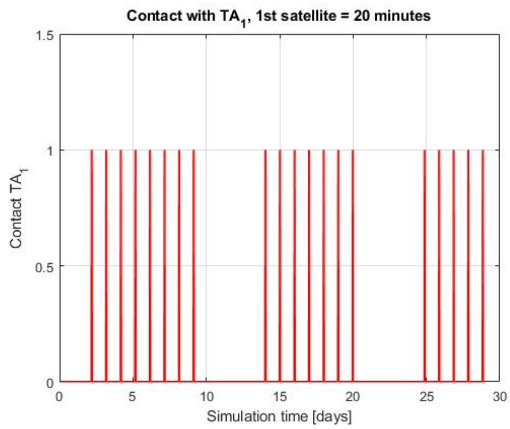



Figure 4.7: Target Area Contacts

The satellite passes by Catalonia every day during periods of 7 days. Then there are periods of 5 days, were the satellite does not pass through Catalonia.

This means that for 11 days, during 7 of them the satellite obtains photos of Catalonia, obtaining a maximum revisit period of 5 days.

The constraint is from 2 to 7 days so the revisit time is achieved.

If a shorter revisit time was desired due to the mission goal, a constellation of satellites with the same orbit design must be used to reduce it.

## 4.5.. Satellite Characteristics

In order to do a real mission, the Cube Cat 3 [29] on-board computer and camera has been implemented.

On-board Computer:

- V=3.3V
- P = 2W
- I = 0.5A
- Work cycle = 100

Data characteristics:

- Image compression = 10;
- Overlapping = 0.05;
- Telemetry traffic = 1 kbps
- Data transmission rate = 0.5 Mbps
- Memory size = 2 GB
- SD card maximum = 75%
- Digitization = 32 bits

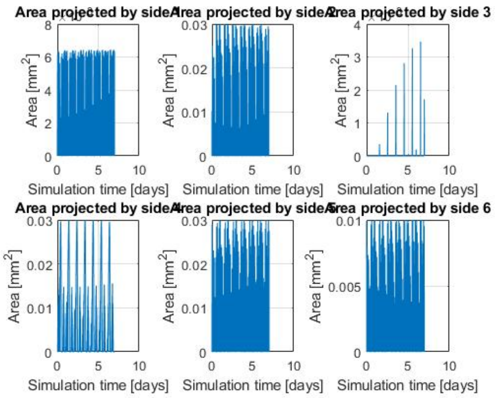
Payload (camera):

- V = 5 v
- I = 2 A
- P = 2 W

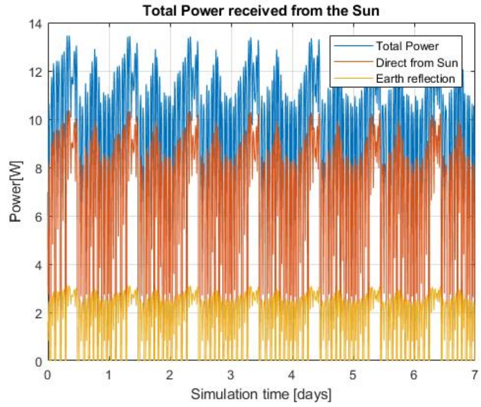
### 4.5.1.. Size of the satellite

As it is a 6U cube, 1U and 2U cubes could not be feasible as not enough energy is acquired. The 2 other available satellite configurations (3U and 6U) have been simulated to check if both are valid under space conditions to work properly.

### 4.5.2.. 3 U Satellite

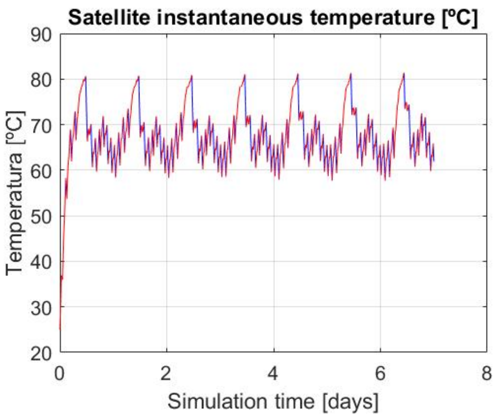



(a) Area projection to the Sun

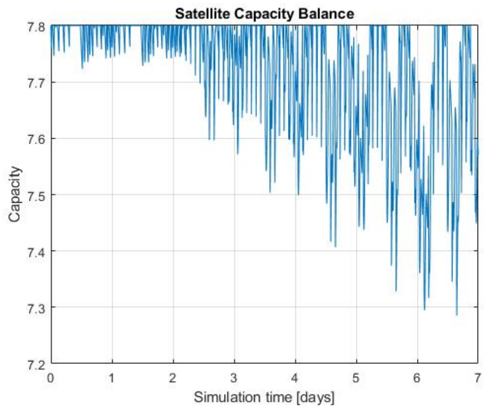



(b) Power absorbed

Figure 4.8: 3U satellite area projection and power absorbed

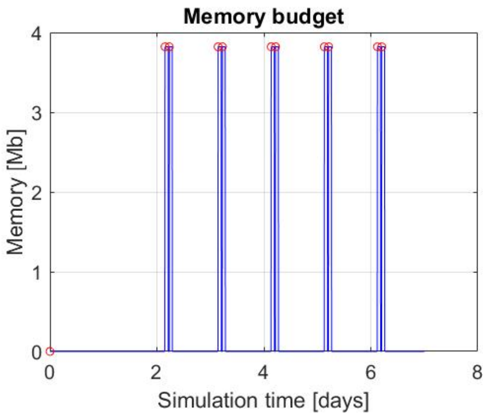



(a) Satellite's temperature

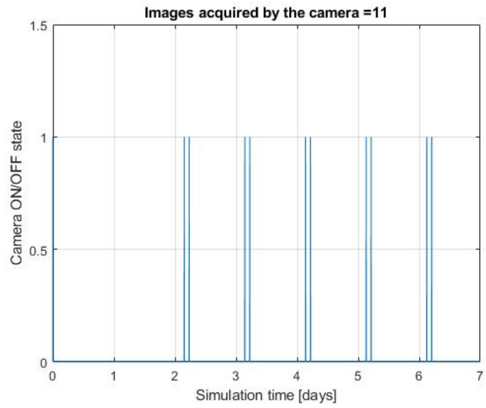



(b) Capacity

Figure 4.9: 3U satellite thermal budget and battery capacity




(a) Memory budget

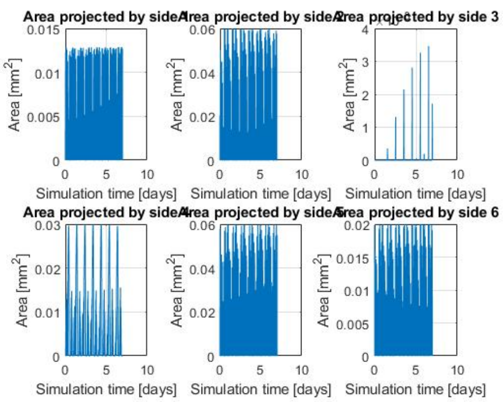



(b) Photos taken by the payload

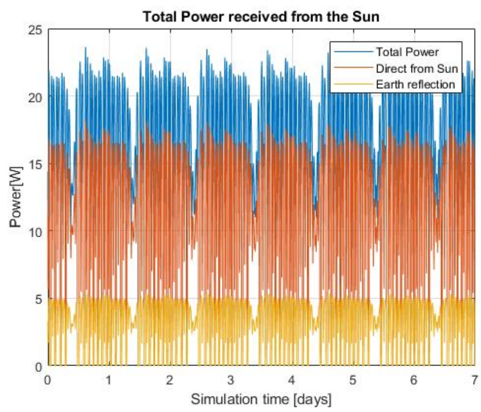
Figure 4.10: 3U satellite memory budget and data collected

It can be observed that the capacity evolution is negative. As more days pass, it is reduced, so in sometime, the capacity won't be enough to make the payload camera work and collect data. The temperature ranges are comprised between good margins, so an extreme temperature situation won't occur. However, this temperature values are quite high for the electronic equipment, so in the next section this will be discussed.

### 4.5.3.. 6 U Satellite

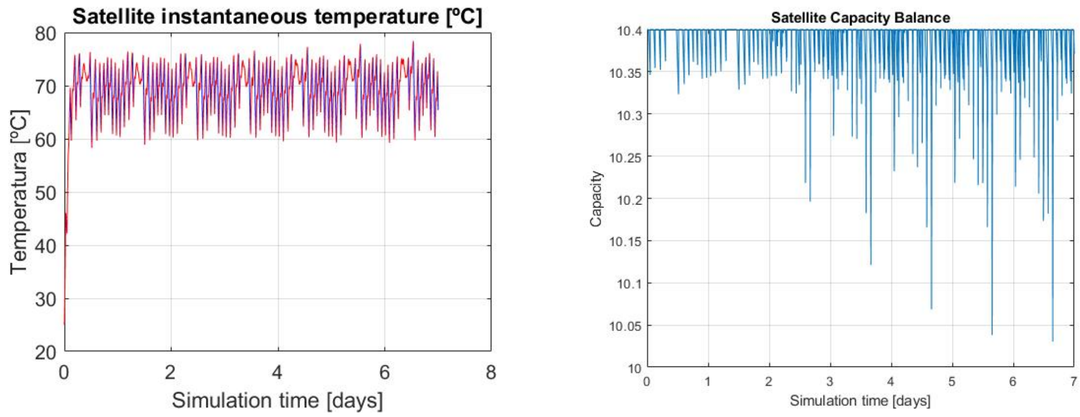



(a) Area projection to the Sun



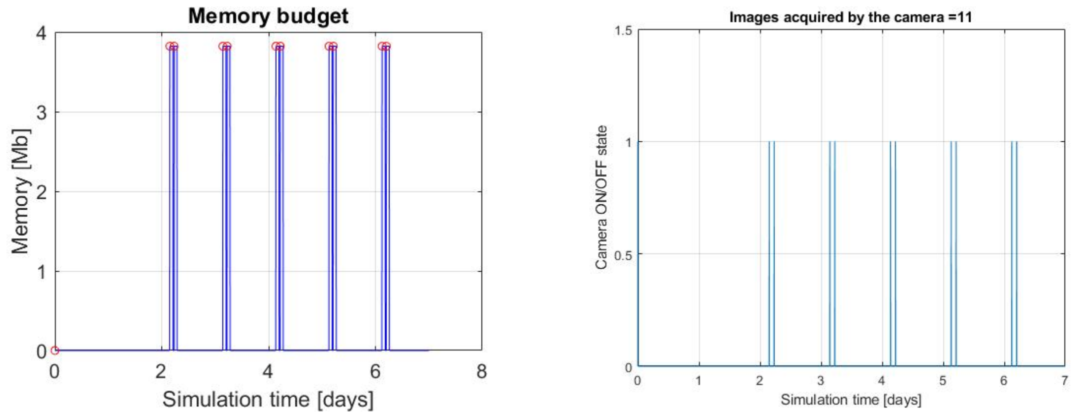

(b) Power absorbed

Figure 4.11: 6U satellite area projection and power absorbed




(a) Satellite's temperature (b) Capacity

Figure 4.12: 6U satellite thermal budget and battery capacity




(a) Memory budget (b) Photos taken by the payload

Figure 4.13: 6U satellite memory budget and data collected

6U satellite is the best option. Bigger size accepts more solar panels and bigger batteries, so that capacity does not fall quickly. As in 3U, capacity evolves negatively as more energy is consumed than acquired. This means that when the capacity reaches certain level, the payload won't make more photos, therefore, some data will be lost. The satellite will enter in sun safe mode to adjust its power balance and then the satellite can work properly.

## 4.5.4.. Painting

The colour of the satellite's painting determines the absorptance and emittance parameters, which describe the percentage of light absorbed and reflected, respectively. This values affect directly to the satellite performance and viability, as they determine its temperature.

Depending on the characteristics of the equipment, a desired margin will be desired to maintain. The simulation done, had the standard values of absorptance and emittance:

| Standard values for absorptance and emittance | | |
|---|---|---|
| | Solar Panel (GaAs) | Chassis (black epoxy) |
| Absorptance | 0.88 | 0.95 |
| Emittance | 0.8 | 0.85 |

Table 4.5: First absorptance and emittance values simulated

In the following figure, the more common surface paintings are shown:

| Surface | Absorptance ($\alpha$) | Emittance ($\varepsilon$) | $\alpha/\varepsilon$ |
|---|---|---|---|
| Polished beryllium | 0.44 | 0.01 | 44.00 |
| Goldized kapton (gold outside) | 0.25 | 0.02 | 12.5 |
| Gold | 0.25 | 0.04 | 6.25 |
| Aluminium tape | 0.21 | 0.04 | 5.25 |
| Polished aluminium | 0.24 | 0.08 | 3.00 |
| Aluminized kapton (aluminium outside) | 0.14 | 0.05 | 2.80 |
| Polished titanium | 0.60 | 0.60 | 1.00 |
| Black paint (epoxy) | 0.95 | 0.85 | 1.12 |
| Black paint (polyurethane) | 0.95 | 0.90 | 1.06 |
| —electrically conducting | 0.95 | 0.80–0.85 | 1.12–1.19 |
| Silver paint (electrically conducting) | 0.37 | 0.44 | 0.84 |
| White paint (silicone) | 0.26 | 0.83 | 0.31 |
| —after 1000 hours UV radiation | 0.29 | 0.83 | 0.35 |
| White paint (silicate) | 0.12 | 0.90 | 0.13 |
| —after 1000 hours UV radiation | 0.14 | 0.90 | 0.16 |
| Solar cells, GaAs (typical values) | 0.88 | 0.80 | 1.10 |
| Solar cells, silicon (typical values) | 0.75 | 0.82 | 0.91 |
| Aluminized kapton (kapton outside) | 0.40 | 0.63 | 0.63 |
| Aluminized FEP | 0.16 | 0.47 | 0.34 |
| Silver coated FEP (SSM) | 0.08 | 0.78 | 0.10 |
| OSR | 0.07 | 0.74 | 0.09 |

Notes: SSM, Second Surface Mirror. OSR, Optical Solar Reflector.

Figure 4.14: Absorptance and Emittance values for several surfaces and finishes

By observing the temperature graph of the 6U satellite shown before, the margins of the satellite are quite stable and not extreme. However values are a little high.

To compare, if the surface is painted with polished titanium, the temperature budget obtained is shown in the following graph:

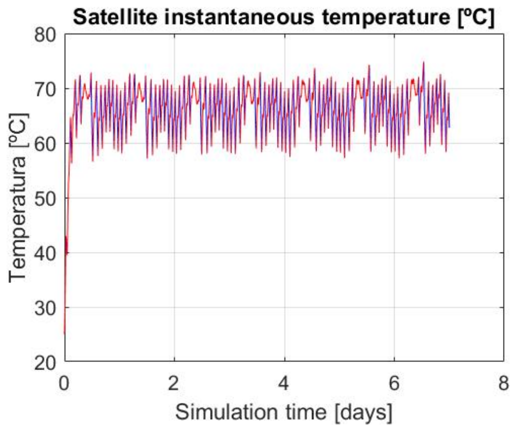



Figure 4.15: Temperature balance with polished titanium surface finish

It can be observed how the temperature values descend around 4$^{o}$C, compared to the black epoxy paint. This is a better surface finish than black epoxy, but it is still high for electronics.

If a more reduced temperature is required, white painting could be the solution:

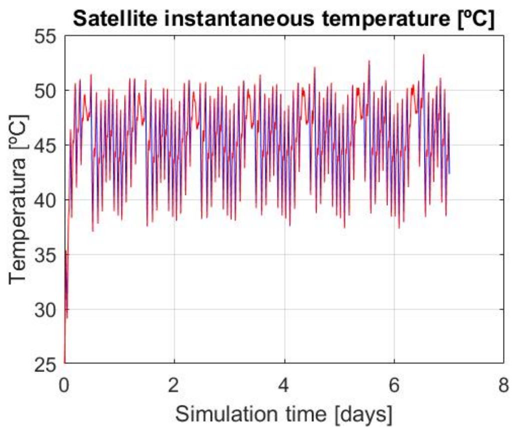



Figure 4.16: Temperature balance with white surface finish

## 4.6.. Final Result

Once a study of the different simulation parts and systems has been done, the following list shows the results obtained for the mission of observing Catalonia.

Target Area - Catalonia

- Latitudes: from 40.5$^{o}$ to 42.9$^{o}$
- Longitudes: from 0.3$^{o}$ to 3$^{o}$

Orbit Design

- LEO
- h = 500km
- Lifetime   7 years
- i = 45.7$^{o}$

Ground Stations

- SPO South Point: Lat = 19.02$^{o}$ Long = -155.67$^{o}$ Alt = 164m
- SGP Singapore: Lat = 1.35$^{o}$ Long = 103.82$^{o}$ Alt = 55m

Satellite Size and Equipment

- 6 U satellite (30x20x10) cm
- EPS P60
- BPX Battery
- 20 P110 Solar Panels

Satellite Painting

- White painting $\alpha = 0.26$, $\varepsilon = 0.83$

This parameters give us a precise approximation about which kind of satellite must be used if the mission goal of observing Catalonia with the constraints mentioned before is desired. Of course, this parameters can vary slightly as it is a simulation software.

# CONCLUSIONS

The idea of this project was to develop a simulator that could take into account as many as real characteristics as possible, to make a more realistic simulator than the first version one. Doing this, means a very large research in theoretical background, as satellite mission analysis needs a very transverse knowledge. Many engineering fields are involved in a satellite, and in order to improve the first version a deep study into all of them had to be done. From the communications or data handling to electronic analysis, passing by orbit characteristics and optimum design.

Moreover, the space environment where the satellite develops the mission, is a very difficult one comparing it to Earth problems. Everything is different and may be taken into account as it affects the satellite's system.

During all the project, the background theory was applied to implement several improvements to the existing version. To do this, a high understanding of the code was also important, as this possible implementations have to be validated by doing several mission simulations, and so, included in the final code. Also real equipment based simulation is necessary to make the satellite parameters more reliable and also, the implementation of 1U 2U 3U sizes may be useful for different users, as many companies are using this nanosatellites for its low costs.

This field of aerospace engineering is constantly growing. Software simulators for satellite mission analysis are very useful for companies of many kinds, that want to put in orbit small satellites, due to its low cost and high utility. Earth observation gives this companies many advantages as allowing them to develop in a more efficient and sustainable way. That is way this kind of software are every day more commonly used and needed.

To conclude with the project, it can be said that more improvements can still be done to the software, as it has lot of different parts. With the pass of the time, as technology improves, many new satellite characteristics could be implemented and so, the mission objectives could be satisfied in a better way, or taking into account lower costs.

The goal of the project has been achieved as a new software version is obtained, more reliable and more accurate with the reality of satellite missions. The validation of the software has been done simultaneously by comparing it to other similar programs, comparing it with the theoretical background studied and obtaining logical values from simulated missions.

The final idea is that the user can use his own satellite parameters and introduce them to the software to obtain a result. By changing the satellite's characteristics a better solution can be found and therefore, reduce costs and improve the mission goals.

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